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Space Programs Summary 37-46, Vol. I

The Lunar Program

For the Period May 1 to June 30, 1967

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Preface

The Space Programs Summary is a six-volume bimonthly publication designed to report on JPL space exploration programs and related supporting research and advanced development projects. The titles of all volumes of the Space Programs Summary are:

Vol. I. *The Lunar Program* (Confidential)
Vol. II. *The Planetary–Interplanetary Program* (Confidential)
Vol. III. *The Deep Space Network* (Unclassified)
Vol. IV. *Supporting Research and Advanced Development* (Unclassified)
Vol. V. *Supporting Research and Advanced Development* (Confidential)
Vol. VI. *Space Exploration Programs and Space Sciences* (Unclassified)

The Space Programs Summary, Vol. VI, consists of an unclassified digest of appropriate material from Vols. I, II, and III; an original presentation of the JPL quality assurance and reliability efforts, and the environmental- and dynamic-testing facility-development activities; and a reprint of the space science instrumentation studies of Vols. I and II.

Approved by:

W. H. Pickering, *Director*
Jet Propulsion Laboratory

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1. *Surveyor* Project

LUNAR PROGRAM

A. Introduction

The *Surveyor* A-21 model flight spacecraft (Table 1) will demonstrate the technique of making soft landings on the moon, extend our knowledge of lunar characteristics and determine the suitability of sites for *Apollo* spacecraft landings.

Surveyor I, the first flight spacecraft, was launched from Cape Kennedy, Florida, on May 30, 1966, and soft-landed on the moon on June 2, 1966. By June 14, when lunar sunset occurred, approximately 100,000 commands had been received by the spacecraft; 10,338 pictures of the spacecraft and its immediate vicinity had been transmitted. During the second lunar day 899 additional pictures were received.

Mission B, which initially gave every appearance of the success of mission A, was terminated in the early morning of September 22, 1966. The spacecraft and all other increments of the mission performed nominally up to execution of the midcourse thrust maneuver. At that point, vernier engine 3 failed to ignite; the resulting imbalance of thrust from vernier engines 1 and 2 imposed a tumbling motion on the spacecraft, from which it failed to recover. Subsequent attempts to fire the vernier engines eventually increased the tumbling rate to 146 rev/min. The main retrorocket then was ignited, and at that time contact was lost with the spacecraft. At 2:53 a.m. PDT, following continuous receiver search for the spacecraft with no results, mission B was terminated.

Because of the high degree of success of the mission up to the point of the midcourse thrust maneuver, data of considerable value were derived. It has already contributed valuable guidance for future *Surveyor* missions.

Table 1. Model and vehicle designations

Designation	Description
SC-1–7	Flight spacecraft with engineering and limited science payload
A-21	Model designation for spacecraft SC-1–7
A-24	Former model designation for spacecraft with proposed scientific payload
T-21	Model A-21 prototype system test spacecraft
T-2	Simplified spacecraft for evaluation of terminal descent propulsion and flight control subsystems
S-2	Test spaceframe for vibration, shock, and static structural tests
S-6, -7	Test spaceframe for vernier propulsion system tests
S-9	Structural test model
S-10	Temperature control model

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Surveyor III was successfully launched from pad 36-B at Cape Kennedy, Florida, April 17, 1967, PST. After being placed in a parking orbit, the spacecraft was injected into a lunar trajectory. Subsequent flight events took place in a normal manner, and a landing was accomplished at a point about 3 mi from the projected landing point in a crater of the Ocean of Storms, April 19.

Despite some early ambiguous telemetry, the spacecraft was in a generally good condition. Following initial TV camera operations, surface sampler experiments were conducted. Four trenches were dug; the largest was 2 in. wide, 10 in. long, and 7½ in. deep. *Surveyor III* also provided the first photographs of an eclipse of the sun by earth.

By the end of its first lunar day, the spacecraft had responded to more than 30,000 commands and returned 6315 pictures. A brief and inconclusive thermal experiment was conducted after sunset, and then all operations were discontinued until dawn of the second lunar day.

Hughes Aircraft Company (HAC), Space Systems Division, is under contract to fabricate the *Surveyor* A-21 spacecraft. The launch vehicle, a combination *Atlas/Centaur,* is provided by General Dynamics/Convair (GD/C). Control, command, and tracking functions for the *Surveyor* missions are performed by the *Surveyor* Mission Operations System and the Jet Propulsion Laboratory (JPL) Deep Space Network.

B. Spaceflight Operations

1. Mission C Operations

The following is abstracted from Section III, "Mission Conduct," of the *Surveyor* Mission C Spaceflight Operations Report (SSD 74071), dated May 1967. This excerpt concerns itself with the transit and initial lunar day operations.

a. Director's report.

Introduction. Surveyor III landed in a shallow crater on the Ocean of Storms within 1.6 mi of the target site. Launched on day 107 at 07:05:01 GMT (11:05:01 p.m. PST, April 16, 1967), the spacecraft was able to achieve a successful landing approximately 65 h later, in spite of hopping twice after initially contacting the surface, because of the failure of the vernier engines to shut off automatically at the proper time. *Surveyor III* has taken thousands of TV pictures (both 200- and 600-line) and has successfully operated its soil mechanics/surface sampler (SM/SS). However, the following paragraphs are limited to a summary of mission events beginning with prelaunch final countdown to the completion of earth/sun acquisition by the planar array and solar panel, a little over 8 h after touchdown.

In general, spacecraft performance analysis and command (SPAC) performance during the mission was excellent and contributed greatly to the overall success achieved. Although certain inflight anomalies, facility support problems, and personnel errors occurred, none was of such a nature to preclude solution in real time. This description, prepared by members of SPAC, presents a comprehensive report of SPAC activities, including real-time analyses and decisions, as well as a summary of spacecraft performance as determined during the conduct of the mission. In addition, significant operational capabilities demonstrated and problem areas encountered are summarized.

*Brief summary of Surveyor III mission.*The highly successful flight of *Surveyor III,* starting from launch pad 36B and culminating in a soft lunar landing in a crater within the Ocean of Storms, was accomplished between April 17 (GMT day 107) and April 20 (GMT day 110). Following a smooth countdown (except for a 51-min hold at T − 5 to analyze the results of special SC-5 and -3 on-stand roll tests) *Surveyor III* was launched at 07:05:01.059 GMT and proceeded smoothly through a $64^h59^m17^s$ flight, with touchdown occurring at 00:04:18 GMT on day 110. The landing of *Surveyor III* was softer than that of *Surveyor I* even though there were three touchdowns, because of the loss of the 14-ft mark; estimated touchdown velocity was 7 f/s. The only confirmed spacecraft malfunction during the flight was the nonreceipt of the 14-ft mark during terminal descent. A lower than expected SM/SS auxiliary electronics temperature is anomalous.

The suspected unequal thrust indications of the vernier engines obtained during midcourse correction were judged, after considerable analysis by SPAC and project management, to be most likely within the error resolution of the *Surveyor* signal processing system.

During the postlanding period, approximately at the time of the second touchdown, the telemetry data became garbled. However, over a period of time, SPAC was able to generate correction factors for the telemetry signals and obtain realistic telemetry indications to accurately

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assess spacecraft performance and accomplish power and thermal management. Good 200- and 600-line TV pictures were obtained whenever the TV system was operated, and the SM/SS performed all tasks to specification.

Table 2 indicates the time at which major milestones of this phase of the mission occurred. In order to accomplish all transit operations, 346 commands were sent to the spacecraft up to touchdown.

During the prelaunch phase, project management requested: (1) that SPAC become totally briefed on the roll actuator data "anomaly" observed in the systems readiness test prior to the countdown, as well as during the special test at Cape Kennedy prior to the start of the launch countdown procedure; (2) directed that SPAC indicate whether they would recommend to launch or to hold based on their assessment of the available data, and (3) directed that SPAC be briefed on the conduct and results of the SC-5 roll actuator test conducted at El Segundo under Cape Kennedy initiation and direction. As a result of (1), SPAC responded to (2) with a hold recommendation, after the count had reached $T - 5$ min. As a result of (3), SPAC recommended to launch. Both the trend and failure analysis (T and FA) and performance analysis (PA) groups participated in the analysis conducted in the SFOF and were aided by good quality Air Force Eastern Test Range (AFETR) data output from the 7044 and 1219 computer systems, including a playback of the El Segundo SC-5 test in near real time. An unnecessary delay in the launch had been introduced, however, since the AFETR and SPAC briefings by El Segundo were conducted in series, with SPAC being briefed last. (The delay worked to some advantage, however, allowing the ship *Twin Falls* to better relay spacecraft data to SPAC covering an anticipated dead period between Ascension and Pretoria tracking stations, even though it was still steaming toward its intended launch destination.)

Injection, separation, sun acquisition, solar panel deployment, DSIF acquisition, initial commanding and interrogations, Canopus verification and acquisition, and premidcourse maneuvers were executed and successfully completed with dispatch.

Shortly after the successful midcourse thrusting, a detailed analysis was conducted by the T and FA group concerning the assessment of spacecraft telemetry indicating level of thrust. After all possible sources of error were corrected for, it was concluded that the apparent

Table 2. *Surveyor III* mission milestones

Event	Date, April, 1967	Time	
		PDT, h, min, s	Time after launch
Launch	16	23:05:01	
Injection		23:38:49	33^m48^s
Separation		23:39:54	34^m33^s
Automatic sun acquisition completed		23:47:58	42^m57^s
Automatic solar panel deployment completed		23:49:54	44^m53^s
Spacecraft visibility at Canberra begins		23:54	49^m
Initial DSIF acquisition (two-way lock) confirmed	17	00:00	55^m
First ground command sequence initiated		00:09:48	$1^h4^m47^s$
Canopus verification started		8:09:12	$9^h9^m11^s$
Canopus acquisition completed		8:27:51	$9^h22^m50^s$
First premidcourse attitude maneuver initiated		20:46:50	$21^h41^m49^s$
Midcourse thrust executed		21:00:02	$21^h55^m01^s$
Sun reacquired		21:04:37	$21^h59^m36^s$
Canopus reacquired		21:08:11	$22^h3^m10^s$
First terminal descent attitude maneuver initiated	19	15:23:30	$64^h18^m29^s$
Retromotor thrust direction properly positioned		15:32:50	$64^h27^m49^s$
Final roll completed		15:36:42	$64^h31^m42^s$
Altitude marking radar mark generated		16:01:13	$64^h56^m12^s$
Vernier engine ignition		16:01:18	$64^h56^m17^s$
Retroengine ignition		16:01:19	$64^h56^m18^s$
Retroengine separation		16:02:14	$64^h57^m13^s$
Radar control initiated		16:02:15	$64^h57^m14^s$
Touchdown (initial)		16:04:18	$64^h59^m17^s$
Touchdown (second)		16:04:42	$64^h59^m41^s$
Touchdown (final)		16:04:54	$64^h59^m53^s$
First 200-line TV picture		17:02:32	$65^h57^m31^s$
Earth/sun acquisition completed	20	00:15:00	$73^h9^m59^s$
First 600-line TV picture		00:42:00	$73^h36^m59^s$

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remaining thrust level discrepancies in received telemetry were probably within the resolution error of the *Surveyor* signal processing telemetry system.

Sun and Canopus reacquisition, terminal maneuvers, and descent were completed successfully, with dispatch, and ignition, retromotor separation, and radar control proceeded normally up to the generation of the 14-ft mark. The latter was not received, the vernier engines did not shut off at touchdown, and touchdown itself became difficult to identify in real time, primarily because the spacecraft did not remain on the lunar surface for more than a moment.

As a result of the unusual telemetry indications (only 3 of the 13 possible functional groups of touchdown telemetry indicators were operable at touchdown), transmission of the "thrust phase power off" command was intentionally delayed by SPAC some 13 s longer than the nominally scheduled Engineering Planning Document-180 sequence of events time. When the telemetry became unintelligible and was confirmed as such, the command was transmitted, the engines shut off, and the spacecraft dropped to the lunar surface. Although it was not known at the time, the spacecraft had actually touched down twice before the ground backup command was transmitted. An added reason for the delay of the thrust phase power-off command was due to the premission concern about nylon expansion; if one or more engines were stuck on, transmission of the command to the spacecraft could have resulted in an increased thrust level and caused liftoff and/or toppling.

This particular type of *Surveyor* landing was analyzed in detail by J. M. Hansen in Hughes Aircraft Company (HAC) interdepartment correspondence 2253.1/655, dated May 24, 1966. The actual landing, reconstructed by SPAC in Fig. 1, is in remarkable agreement with Hansen's original analysis.

With the receipt of "bad data" from the spacecraft on the moon, SPAC worked to solve what appeared in real time to be: (1) a signal processing anomaly and/or (2) a power subsystem anomaly. Twenty-one separate sequences consisting of 87 commands were transmitted to the spacecraft during the first 55 min after touchdown, with all but six being "nonstandard" sequences aimed at solution of the problem (Table 3). Excluding antenna solar panel positioning (A/SPP) and TV, this command rate, although not staggering, is greater than any standard

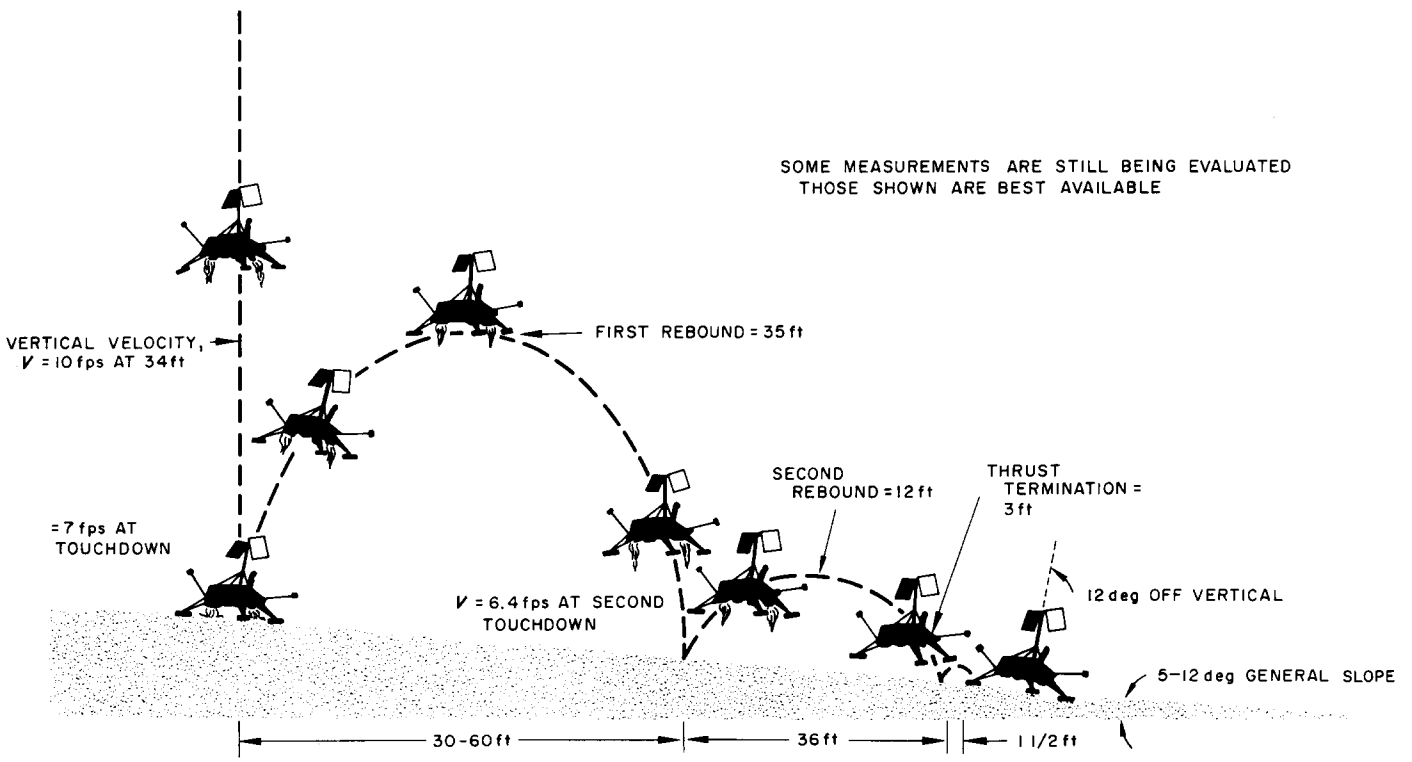

Fig. 1. *Surveyor III* landing profile

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mission operational phase, including "active" phases such as midcourse and terminal descent, and, as such, represents an operational milestone.

The standard sequences consisted of: (1) routine turnoff of strain gages and accelerometer amplifier; (2) normal turnoff sequence for thrust phase power; and (3) four project-directed sequences to prepare the spacecraft for 200-line TV pictures by going back to 1100 bits/s and high power, and turning summing amplifier and engineering commutators off.

At approximately 6 min after touchdown, it was suspected that all digital words were transmitting correct and accurate data. Approximately 30 min after touchdown, it was decided that the failure mode was identifiable via SPAC nonstandard procedure (NSP) No. 27, signal processing, point 6A, and was definable as follows:

(1) All digital on-off indications in all digital words were confirmed normal for a healthy spacecraft.

(2) TV ID signal data should be normal (confirmed in first TV sequence).

(3) TV data should be normal (confirmed in first TV sequence).

(4) The bootstrap disconnect switch in the master switch for the TV auxiliary failed in the short mode.

A conflict concerning whether to prepare for TV or work the power anomaly was quickly resolved, and it was decided to work the power problem after the TV pictures were obtained. The 200-line TV sequence consumed another 34 min and the spacecraft was then made available to SPAC to continue investigating the anomaly. A quick final analog-to-digital converter check of the signal processing anomaly was made in parallel with a spacecraft shutdown and turnon sequence intended to clear the power anomaly, if any. An additional 25 separate sequences consisting of 49 commands were transmitted to the spacecraft during the next 87 min with all but 5 being nonstandard sequences aimed at a solution of the anomaly (Table 3). The 5 standard sequences consisted of a routine turnon of the TV and SM/SS heaters and four project-directed sequences to return to high power, 1100 bits/s, and TV.

Just prior to returning to 1100 bits/s, T and FA noted that some half-dozen power analog signals were apparently indicating properly for a healthy spacecraft. At 1100 bits/s these signals went bad, and the signal processing anomaly took on further definition, having unexpected capacitive effects. During the second and third TV sequences, some 3 to 4 h after touchdown, it was decided in a meeting with SPAC, space science analysis and command (SSAC), and project management to (1) continue assuming a healthy spacecraft and (2) acquire the sun and earth as quickly as possible with A/SPP as soon as SSAC 200-line TV objectives were completed.

The second TV sequence consumed approximately 50 min; the spacecraft was returned to SPAC for a short check of commutators 4 and 5 and then returned again to SSAC after only 20 min for the third and final 200-line TV sequence, lasting another 40 min.

Consequently, 5 h after touchdown SPAC monitored data at 137.5 bits/s for a little more than an hour to gain confidence in the power analog data. At 6 h and 28 min after touchdown, sun and earth search were finally initiated. Earth was acquired with a planar array first sidelobe at 7 h and 19 min after touchdown, some 50 min later. The A/SPP was finally (for day 110) positioned on the sun with the main planar array lobe peak on the earth at 8 h and 12 min after touchdown. Published premission SPAC estimate of earth and sun acquisition was 75 min maximum for a level spacecraft, utilizing analog telemetry signals, neither of which was the case here.

Operational capabilities. In addition to confirming performance parameters of the spacecraft, Mission C was instrumental in further demonstrating unique operational capabilities. Some of the more impressive of these, in addition to those compiled for *Surveyor I* and again demonstrated by *Surveyor III*, are

(1) Steady, consistent, and methodical sending of command sequence after command sequence (21 in the first 55 min) after touchdown, upon receipt of essentially unintelligible telemetry data, in order to work the apparent anomaly and to protect the spacecraft. This was made possible by real-time operational personnel experience, training, and discipline and the highly flexible and efficient *Surveyor* command link designed for just such a situation, using man in a decision loop to control and evaluate spacecraft performance. The command rate for these nonstandard extemporaneous sequences was more than 1½ per min, which exceeds the rate for *Surveyor* during active periods, such as midcourse correction and terminal descent.

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Table 3. Summary of posttouchdown SPAC operations[a]

	Item	GMT[b]
	Up to start of first TV sequence	
1	Two thrust phase power-off commands	00:04:51
2	Four thrust phase power-off commands	00:06.52
3	Switch from analog-to-digital-1 (A/D) converter 1 to A/D-converter 2	00:09:27
4	Two radar altimeter and doppler velocity sensor (RADVS) off followed by two flight control-off commands	00:10:35
5	Touchdown strain gages and channels, propulsion strain gages, and auxiliary accelerometer amplifiers off	00:12:08
6	Switch from command 6 to command 5	00:13:31
7	Switch from command 5 to command 2	00:18:46
8	RADVS off	00:21:21
9	High current mode off and return to main battery mode	00:22:03
10	Reduce bit rate from 1100 to 550 bits/s	00:22:34
11	Switch transmitter B to low power, off	00:26:30
12	Reduce bit rate from 550 to 137.5 bits/s	00:27:58
13	Switch from transmitter B (off) to transmitter A (with a sum amplifier) on omnidirectional antenna B	00:34:56
14	Two thrust phase amplifier power-off commands	00:40:17
15	Two RADVS-off followed by two flight control-off commands	00:40:34
16	Switch transmitter A to high power	00:49:47
17	Increase bit rate from 137.5 to 1100 bits/s	00:50:35
18	Commutator 4 off	00:55:17
19	Sum amplifiers off	00:55:32
20	Bypass main overload trip circuit twice	00:58:08
21	Ten RADVS-off commands	00:59.00
22	Start 200-line TV	01:00:58

	Item	GMT
	After first TV sequence and up to second TV sequence	
1	Come up in commutator 5 high power	01:34:48
2	Compartment A heater off	01:42:48
3	Switch from commutator 5 to commutator 4	01:48:29
4	Switch from A/D converter 2 to A/D converter 1	01:54:36
5	Transmitter A to low power and filament off	02:00:51
6	Optimum charge regulator off (0310)	02:01:19
7	Engineering commutators off	02:02:08
8	A/D converter power off	02:02:20
9	33, 7.35, 3.9 kHz subcarrier oscillators off	02:02:31
10	Summing amplifiers off	02:02:43
11	Transmitter low-power off	02:02:54
12	Nonessential loads off (interlocked)	02:03:06
13	Enable main overload trip circuit	02:05:13
14	Transmitter B to planar array (0116)	02:05:37
15	Transfer switch A to low power (0126)	02:05:42
16	Transmitter A and narrow band voltage controlled crystal oscillator, no modulation	02:05:47
17	Phase summing amplifier A on	02:14:53
18	Come up 137.5 bits/s with A/D-1 converter 2	02:15:07
19	Presumming and 150 amplifier on	02:15:55
20	Commutator 4 on	02:16:22
21	SM/SS, vidicon, and TV electronics temperature control on	02:27:57
22	Switch from command 4 to command 5	02:31:40
23	Switch to high power	02:57:45
24	Switch from 137.5 to 1100 bits/s	03:01:05
25	Prepare for TV, command, and summing amplifier off	03:02:57
26	Start 200-line TV, first lunar phase TV	03:07:47

[a]Touchdown = 110:00:04:18.
[b]Time is start of configuring for function listed.

(2) Remarkable positioning of the solar panel essentially on the sun and the planar array rather precisely on the earth in only 100 min without possibility of reference to *Surveyor* telemetry signals (or any other *Surveyor* output for 50 min until earth acquisition) with the spacecraft in a lunar crater on an unknown slope of some 12.5 deg.

(3) Ability of the SPAC T and FA group to come up with calibration fixes to practically all telemetry signals, for all commutators, at 17.2 bits/s some 144 h after touchdown, allowing accurate assessment of spacecraft performance throughout *Surveyor III's* first lunar day operations.

(4) Ability of the SPAC real time performance analysis group to come up with thermal channel correction factors based on *Surveyor I* experience some 48 to 130 h after touchdown, which provided the first thermal data interpretation breakthrough of the first lunar day operations. The first proof of the assumed correction factors was obtained some 96 h after touchdown, when transmitter frequency was utilized to verify compartment temperature.

(5) Presence and performance of the new T and FA group within SPAC operations. Studies were accomplished which provided a firmer base on which to justify real-time decisions and could have been the basis for reversing or changing real-time decisions, if necessary. In particular, the prelaunch analysis of the roll actuator anomaly, the midcourse anomaly relative to uneven engine thrust levels, and the posttouchdown analysis that resulted in updated calibration coefficients were detailed

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studies that real-time operations could not have found the time to accomplish. The last-named analysis was one of the major contributions to accurate performance assessment of the spacecraft after touchdown.

Significant deviations from EPD-180. Significant changes in procedure from the prepared standard mission sequence documented in EPD-180 are summarized in Table 4. Also included in this table are the reasons for each deviation. Briefly, the sequence change resulted from

(1) DSS report that the modulation of the subcarrier on the carrier was not normal, resulting in a special nonstandard sequence for investigating this problem. (Report proved to be incorrect.)

(2) Desire to obtain maximum possible sampling rate during star verification, resulting in selection of 4400 bits/s for this sequence.

(3) Additional gyro drift checks obtained during coast phases I and II to obtain the best estimate of the drift of each gyro for use in the terminal descent computations.

(4) Normal acquisition of the sun and Canopus following the reverse midcourse maneuvers, resulting in a decision by the space flight operations director to eliminate the postmidcourse star verification sequence.

(5) Reduction in bit rate from 550 to 137.5 bits/s, caused by bit error rate exceeding the desired 3×10^{-3} rate.

(6) Request of flight path analysis and command group (FPAC) to perform an additional voltage controlled crystal oscillator frequency check, resulting in an additional check approximately 8 h prior to touchdown.

(7) Decision to combine the last two preterminal descent engineering interrogations.

(8) Additional sampling of mode-6 data.

(9) Delay in commanding the survey camera vidicon temperature control on until mode 4 was selected to permit verification that the temperature was greater than specified (i.e., $> -20°F$) for turnon.

Operational decisions and changes made during mission. In addition to accomplishing its normal function of verifying spacecraft status during and between all normal command sequences, the Performance Analysis group was responsible for making many real-time analyses that resulted in decisions and recommendations for amending or implementing the sequences documented in EPD-180-S/MC. Furthermore, ground equipment and station influences on the sequence of events were iterated from the station manager/*Surveyor* operations chief position to SPAC via the command control line. In addition, many outputs were provided for other operational elements of the SFOF (such as FPAC, the spaceflight operations direction, mission control panel, and the DSIF stations) to provide these organizations with required data.

In addition to some 79 decisions and outputs based on standard spacecraft performance, which were summarized in Table 3-2 of the SPAC Mission A Report (SSD 64189R), Table 5 summarizes the significant decisions, recommendations, and outputs given by SPAC during Mission C.

b. First lunar day.

Immediately after first apparent touchdown, PCM telemetry remained in lock and indicated that the vernier engines were still thrusting and the spacecraft was still in motion. Commanding flight control thrust phase power off was deferred so as not to perturb the flight control stabilization loop.

All subsystems reported erratic changes in their telemetry. One of the first indications of a signal processing problem was the fact that observed temperatures shifted unrealistically fast by 10°F or more. Subsequent investigation showed that the lower the bit rate, the more plausible were the data. Also, a set of applicable conditions for this time was contained in NSP 27, point 6A. The electrical power signals remained nearly constant at a critical level, lending support to the suspicions of a signal processing problem.

The NSP point 6A could not be ratified, however, until the SM/SS was turned on, with the expectation that it would inhibit a suspected faulty master switch in signal processing. Soil mechanics/surface sampler turnon had to be delayed until it was predicted to be thermally ready for operations ($\cong$ touchdown (TD) + 10h initally, TD + 30h subsequently.)

Meanwhile, solar panel positioning was accomplished by maximizing optimum charge regulator (OCR) output current, since the standard signals used for positioning the secondary sun sensors were not immediately interpretable.

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Table 4. Summary of significant deviations from EPD-180

Mission phase	Time after launch	Event/deviation	Reason for deviation
DSIF acquisition through star acquisition	6^h2^m	Unscheduled sequence for switching to 4400 bits/s and returning to 1100 bits/s to determine if a modulator problem existed	DSS 42 reported that sideband energy in received signal was down by approximately 10 to 11 dB from carrier (instead of 5 to 6 dB expected). It was thought that possibly modulation index was incorrect.
DSIF acquisition through star acquisition	6^h27^m	Bit rate reduction from 1100 to 550 bits/s executed early	To check modulation loss with a different subcarrier oscillator and bit rate.
DSIF acquisition through star acquisition	7^h2^m	Unscheduled commanding analog-to-digital converter power off and on implemented	To check sideband energy in unmodulated subcarrier.
DSIF acquisition through star acquisition	8^h58^m	Unscheduled bit rate increase to 4400 bits/s for star verification	To obtain a greater sampling rate of Canopus sensor signals during star verification.
	9^h31^m	Unscheduled bit rate decrease to 1100 bits/s following star verification	To return spacecraft to prestar verification status.
Coast phases I and II		Gyro drift checks run as follows:	Measured gyro drift during first measurement was excessive. Accurate value was desired so that drift could be compensated for during retroengine maneuvers. Only four drift checks are scheduled, but others were felt to be desirable to increase confidence in measured values.
	10^h24^m	(1) Start gyro drift, all axes	
	12^h12^m	(2) End drift	
	12^h19^m	(3) Start drift, all axes	
	14^h37^m	(4) End drift	
	16^h11^m	(5) Start drift, roll only	
	19^h6^m	(6) End drift	
	24^h31^m	(7) Start drift, all axes	
	26^h39^m	(8) End drift	
	34^h26^m	(9) Start gyro drift, all axes	
	36^h27^m	(10) End pitch and yaw drift, continue roll	
	36^h32^m	(11) Start drift in pitch and yaw, continue roll	
	37^h18^m	(12) End drift	
	37^h22^m	(13) Start gyro drift, all axes	
	39^h45^m	(14) End drift	
	39^h49^m	(15) Start gyro drift, roll only	
	42^h11^m	(16) End drift	
	42^h19^m	(17) Start gyro drift, all axes	
	44^h32^m	(18) End drift	
	44^h52^m	(19) Start gyro drift, roll only	
	47^h40^m	(20) End drift	
	47^h44^m	(21) Start gyro drift, all axes	
	50^h04^m	(22) End drift	
	51^h16^m	(23) Start gyro drift, roll only	
	53^h36^m	(24) End drift	
	55^h23^m	(25) Start gyro drift, all axes	
	57^h51^m	(26) End drift	
	58^h1^m	(27) Start gyro drift, roll only	
	61^h46^m	(28) Start gyro drift, all axes (sent unintentionally)	
	61^h51^m	(29) End drift	

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Table 4 (contd)

Mission phase	Time after launch	Event/deviation	Reason for deviation
Coast phase II	22^h10^m	Unscheduled bit rate reduction from 4400 to 1100 bits/s	Signal strength sufficient to sustain 1100 bits/s rather than 550 bits/s planned.
Coast phase II	25^h30^m	Postmidcourse star verification and acquisition not performed	Postmidcourse reverse maneuvers resulted in reacquisition of sun and star.
Coast phase II	49^h00^m	Vernier fuel tank 2 and vernier oxidizer tank 3 thermal control not turned on	Vernier oxidizer tank 2 temperature only low one.
Coast phase II	50^h52^m	Unscheduled bit rate reduction from 550 to 137.5 bits/s	Signal strength lower than assumed in EPD-180 sequence, resulting in bit error rate approximately 9×10^{-3}.
Coast phase II	56^h55^m	Special voltage controlled crystal oscillator (VCXO) frequency check	Request by FPAC to expedite execution of VCXO check preceding terminal descent.
Coast phase II	60^h00^m	Engineering interrogation combined with subsequent interrogation	Per space flight operations director instruction, to save time and decrease operational activity near terminal descent.
Coast phase II	60^h38^m	Unscheduled sampling of mode-6	To evaluate mode-6 data.
Coast phase II	62^h27^m	Modified command sequence for entry and exit of gyro speed check	To accommodate current pulse code modulation bit rate of 137.5 bits/s
Coast phase II		Delayed commanding of survey camera vidicon temperature control on until mode 4 selected in next engineering interrogation	So that proper temperature could be verified prior to commanding on temperature control.
Coast phase II	64^h05^m	Modified sequence for bit rate increase	Due to operation at 137.5 rather than at 550 bits/s.

Table 5. Summary of significant real-time decisions/output

Mission phase	Time (T) before and after Launch (L)	Decision/Output	Reason
Prelaunch countdown	$T-40^m$ $T-20^m$	AFETR $T-40^m$ and $T-20^m$ reports analyzed and checked. Predictions provided to FPAC for transmission to DSS 42.	Required by DSS 42 for setting DSS receiver and transmitter frequencies properly to ensure expeditious achievement of initial DSS acquisition.
Prelaunch countdown	$T-5^m$	Confirmation of spacecraft status and readiness for launch. Launch held while SPAC evaluated roll actuator problem per SFOD direction.	Provided for automatic checkout equipment (ACE-1) visibility of spacecraft condition and resolution of roll actuator problem.
DSS acquisition through star verification	$L+1^h10^m$	Decide to delay command cruise mode on, since star error signal was fluctuating along with high star intensity signal.	To conserve gas by keeping roll axis control in inertial mode.
DSS acquisition through star verification	$L+1^h15^m$	Computed dB margin for 1100 bits/s transmission with transmitter low power. Advised bus chief that there was sufficient margin at this time, so that this mode of operation could continue at least until star verification and acquisition.	To determine capability of continuing to transmit spacecraft data at 1100 bits/s during coast phase I.
DSS acquisition through star verification	$L+1^h22^m$	Recommend that transponder A not be turned on.	Requirement of ± 25 kHz automatic frequency control (AFC) in receiver A was satisfied with zero phase error indications in receiver B.

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Table 5 (contd)

Mission phase	Time (T) before and after Launch (L)	Decision/Output	Reason
DSS acquisition through star verification	L+6^{h}02^m L+7^{h}02^m	Prepared and executed nonstandard sequences for evaluation of DSS reported low received sideband power.	Reported received sideband power was lower than predicted.
DSS acquisition through star verification	L+6^{h}24^m	Prepared briefing for mission control group, including brief review of performance of each spacecraft subsystem up to this point, review of present SPAC status board parameters, plots of actual versus predicted values for remaining battery energy, transmitter temperatures, compartment-A tray temperatures, compartment-B tray temperatures, received signal strength, etc.	To establish policy for midcourse attitude maneuvers and correction.
Coast phase I	L+7^h to L+8^h	Participated in preliminary midcourse policy conference with mission control and FPAC personnel. Decision reached was to execute the midcourse correction at 20^h rather than 40^h.	To establish ground rules for finalizing midcourse correction analyses.
Coast phase I	L+7^{h}55^m	Telecommunication analysis for Canopus verification/acquisition sequences performed. This study indicated that during roll there would be margins of 1 $\pm$5 and 15 $\pm$5 dB for uplink command and carrier signals, and $-$18 $\pm$5 and $-$13 $\pm$ 5 dB for downlink telemetry and carrier signals, using high-powered transmitter, omniantenna-B operation, with a data rate of 4400 bits/s.	Operational sequence required to acquire Canopus.
	L+7^{h}55^m	Recommendations for Canopus acquisition sequence were: (1) use omniantenna B; (2) use high power only during roll maneuver (do interrogation in low power); (3) go one-way lock; (4) only one roll required, since it was predicted that no significant omniantenna B nulls would be encountered; (5) use coast mode commutator instead of mode 1 to get receiver A automatic gain control data so that information could be obtained. Prepared sequence for implementing recommendations.	
Coast phase I	L+9^{h}20^m	Verified star map by identifying three stars and one planet plus Canopus and earth. Confirmed that star lockon would be achieved on proper star (Canopus).	To establish vehicle orientation to known space coordinates so that midcourse correction could be executed properly.
Coast phase I	L+9^{h}21^m	Decided to utilize automatic star acquisition mechanism.	Results of star map indicated sufficient Canopus intensity for automatic star acquisition.
Coast phase I	L+10^{h}15^m	Reported SM/SS auxiliary electronics temperature below normal.	Auxiliary electronic temperature was below anticipated; indications were that heaters were not cycling.
Coast phase I	L+12^{h}19^m L+16^{h}11^m	Recommended initiation of additional gyro drift check.	To obtain as much drift data as possible prior to midcourse. As drift check at 10^{h}24^m indicated, roll and yaw drift rates were out of specification.

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Table 5 (contd)

Mission phase	Time (T) before and after Launch (L)	Decision/Output	Reason
Coast phase I	L+14^{h}30^m	Briefed ACE-1 on midcourse maneuver telecommunication analysis. Recommended roll-yaw maneuver pair, using omniantenna B. Eight maneuver pairs were analyzed. Minimum margins of 15.5 and 20.5 dB for downlink telemetry and carrier, and 23.5 ±5 and 37.5 ±5 dB for uplink command and carrier signals during recommended maneuvers.	Prebriefing for final midcourse policy conference with FPAC and mission control.
Coast phase I	L+21^{h}15^m	Recommended delay of postmidcourse star verification.	Canopus was in star sensor field of view after postmidcourse roll.
Midcourse	L+21^{h}56^m	Quick-look evaluation of midcourse thrusting duration performed for mission operations director. Analog recorder indicated duration of burn was 4.3 vs 4.275 s commanded (well within accuracy of recorder timing).	To provide quick estimate of satisfactory execution of midcourse correction.
Coast phase II	L+22^{h}10^m	Reported midcourse data indicated unequal thrust levels during midcourse even though spacecraft was stable.	Possible insufficient moment control during terminal descent.
Coast phase II	L+22^{h}12^m	Evaluated effect of 56-min high-power operational period during midcourse on critical spacecraft temperatures. Results indicated ihat all temperatures were well below allowable upper limits.	For verification of rate of thermal rise during high power operation for use in terminal descent/postlanding planning.
Coast phase II	L+24^{h}31^m	Recommended initiation of additional gyro drift checks.	To obtain more gyro drift data in order to better define the drift rates to be compensated for in terminal descent.
Coast phase II	L+47^{h}08^m	Decided that both main and auxiliary battery were performing satisfactorily, and that both batteries could be put on line.	To determine manner in which both batteries would be sharing load when placed directly on line.
Coast phase II	L+50^{h}30^m	Prepared briefing for preliminary terminal descent conference with mission control and FPAC, including telecommunication, power, thermal, and postlanding-A/SPP-operation analyses. Recommended that if three maneuvers were to be considered, first should be initiated 38 min before retroengine ignition (instead of 33 min).	To obtain preliminary policy decision for final terminal descent computations. Additional time requested for ensuring that operational sequences could be completed on time.
Coast phase II	L+50^{h}52^m	Reduced spacecraft bit rate to 137.5 bits/s.	Data quality (high bit error rate) was unsatisfactory.
Coast phase II	L+58^{h}45^m	Participated in final terminal descent policy conference. Recommendations: (1) use yaw-pitch-roll (three maneuver) sequence and omniantenna B; (2) start at 38 min prior to retroengine ignition; (3) attempt to obtain 1100 bits/s PCM data plus touchdown strain gage data; (4) make decision on attempting to obtain strain gage data based on received signal level of −127.2 dBmW (do not try if signal level <-127 dBmW), after terminal maneuvers are completed; and (5) turn off transponder prior to start of maneuvers.	To obtain terminal descent recommendations.

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Table 5 (contd)

Mission phase	Time (T) before and after Launch (L)	Decision/Output	Reason
Coast phase II	L+59^{h}09^m	Determined auxiliary and main batteries were satisfactory for terminal descent.	Both batteries required on line for terminal descent.
Coast phase II	L+59^{h}30^m	Reported estimated sun (in spacecraft coordinate after touchdown), DSIF received signal strength after touchdown, dB margin for 200-line TV survey camera duty cycle after touchdown.	To facilitate posttouchdown operations.
Coast phase II	L+62^h	Provided to FPAC following gyro drift rates, as result of ten drift checks: (a) Roll +1.1 deg/h (b) Yaw −0.8 deg/h (c) Pitch +0.6 deg/h as well as retroengine burn time of 41.09 s.	For compensating for gyro drift during terminal descent.
Coast phase II	L+63^h	Recommended that DSS 11 receiver be offset during retroengine firing and touchdown phases and coordinated with Deep Space Net (DSN) chief to ensure proper implementation.	To account for expected doppler shifts during retroengine firing and touchdown phases.
Terminal descent	L+64^{h}33^m	Confirmed that adequate signal level was available to permit 1100 bits/s PCM data plus touchdown strain gage data to be obtained (based on −124 dBmW signal level following terminal roll maneuvers), and recommended turnon of touchdown strain gage data channels during vernier descent.	Confirmation for ACE-1 in accordance with final terminal descent policy conference.

Initial planar array positioning was based on the pretouchdown predicted direction of the earth, assuming no spacecraft tilt. The first lobe that was peaked out corresponded in magnitude to a first sidelobe. Subsequent stepping crossed a null into a second sidelobe peak. Reversal of stepping through these two sidelobes led to the mainlobe, which was approximately peaked. Initial estimate of spacecraft tilt was ≅ 11.4 deg, direction 29 deg N of W (later refined to 12 deg, direction 6.5 deg N of W). Six hundred-line TV followed for ≅ 1½ h to end of the Goldstone pass.

Engineering troubleshooting was resumed with all OCR states commanded to determine effect on power signals. A bits/s rate of 17.2 was commanded for the first time, and data at this rate appeared closest to realistic values.

The planar array position was refined, and 600-line TV began soon after the next Goldstone rise (day 111). A policy of using the 17.2 bits/s rate for engineering interrogations between TV sequences was begun at this time. Soil mechanics/surface sampler was turned on at ≅ Touchdown + 32h, but this did not resolve the problem as anticipated. A modified signal processing optional sequence (OS) 17 was performed whereby data were obtained via all commutators, each at all bit rates except 4400 bits/s.

Several plans were proposed to establish one or two point calibrations of the PCM data. One was turnon of vernier line-1 heater and observe its indicated on-off cycle points. (These points were well established in transit.) If cycling would not occur at this time, it would probably occur during the expected lunar eclipse.

Another plan, which was later implemented, involved correlating a lunar spacecraft transmitter narrow-band voltage controlled crystal oscillator frequency to an unknown compartment-A upper transmitter temperature (V-15) via the frequency-temperature curve used by DSS 51 in reverse order of the parameters to aid in initial spacecraft acquisition. The result was a V-15 value ≅ 30°F higher than the data indicated (at 17.2 bits/s, V-15 indicated ≅ 50°F).

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Finally, during the ensuing eclipse, one thermal switch in compartment-B opened at ≅ + 10°F (indicated), 30°F below its nominal open-close set point.

Radio frequency assessment (OS 16) and signal processing assessment (OS 17) were performed several times through the lunar day.

The bit-error rate test (experiment 17) was performed via DSS 42 and DSS 61, while planar array gain/temperature sensitivity (experiment 16) was performed at planar array temperatures of 206 and 110°F.

The signal processing problem was significantly minimized with the establishment of correction curves for temperatures in commutators 4 and 5 at 17.2 bits/s.

Helium was dumped at the direction of the space flight operations director, although the landing legs remained unlocked through spacecraft shutdown.

At spacecraft shutdown (May 4, $00^h02^m43^s$, 1 h after day/night terminator), the battery charge level was approximately 150 A-h. Only one thermal switch in compartment B had opened when the compartment reached 0°F (criterion for spacecraft shutdown), and no switches opened in compartment A (≅ + 17°F), motivating a revision in predicted compartment and battery thermal characteristics through the lunar night and second lunar day.

On day 114 GMT, the deep space net performed antenna boresight and pattern tests at DSS 11 and 42.

Special science thermal experiments were performed on day 116 GMT with excellent results and were terminated earlier than scheduled.

In order to assess the spacecraft for possible touchdown damage, SPAC requested that some 50 specific pictures be taken of the auxiliary battery cable area and the spaceframe (hemispherical survey) from approximately legs 2 through 3 and nearly to leg 1. Although the pictures (approximately six) obtained by SSAC as a result of this request were poor because of sun position, no damage was readily evident.

Mission control conducted, in conjunction with DSS 11, a voice relay experiment with *Surveyor III* on day 123 GMT and encountered somewhat high signal-to-noise ratios.

2. *Surveyor III* Second Lunar Day Revival Attempts

Surveyor III revival activities after the first lunar night consisted of execution of NSP-32, points 1 through 12 and 46 through 57, from 20:09 GMT on day 143 through 23:30 GMT on day 146.

a. Spacecraft status. The attempts to reactivate *Surveyor III* started May 23 at 21:09 GMT, utilizing the procedure defined in the *Surveyor* NSP-32. Table 6 lists these attempts made up to 00:30 GMT on May 25. These points were selected since no transfer switching would be required to check out both transmitters, thereby eliminating commands with heavy current surge requirements from the power system. During the last few hours of day 146, several additional nonstandard power sequences were transmitted to the spacecraft to ensure:

(1) That the optimum charge regulator (OCR) was indeed on by manually commanding it on rather than assuming that it came on automatically.

(2) That the main battery mode was on by manually commanding it on rather than assuming it was on.

(3) That spacecraft turnon was also attempted with the overload trip circuit bypassed.

No RF transfer switching was attempted until day 151. NSP-32, points 1 through 12 and 46 through 57, were repeated on days 147, 148, and 149, using Goldstone passes only, i.e., approximately 12 h/day. No revival attempts were made on day 150. On day 151 at 09:18 GMT, spacecraft revival procedures were initiated utilizing points 1 and 2 of NSP-32 and including the exercise of all feasible power system states (i.e., OCR on, OCR bypass, disable battery transfer logic, restore main battery

Table 6. *Surveyor III* reactivation attempts up to May 25, 1967

Day	Start			Stop		DSS
	Point	GMT	Transmitting system commanded	Point	GMT	
143	1	21:09	A and B	12	23:56	61
144	1	01:01	A and B	12	03:37	61
	1	05:05	A and B	9	10:40	11
	10	12:38	B	12	14:17	42
	42	14:24	A	56	19:08	42
	2	19:07	B	12	22:10	42 and 61
145	46	22:19	A	57	00:11	61

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modes, etc., and combinations of states). Transfer switch command 0117 (transmitter A to planar array antenna) was exercised.

On day 152, antenna transfer switching was finally initiated in earnest in conjunction with NSP-32 as follows:

(1) Command modulation off; then sweep ground transmitter f_0 +20 kHz returning to f_0.

(2) Point 1, then 3261 of point 2 (transmitter A to planar array antenna).

(3) Repeat 1; then command modulation on.

(4) 3262 of point 2.

At this time, a signal was detected at −168 dBmW which was later confirmed to be a sideband of *Lunar Orbiter's* signal in the low side of the *Surveyor* passband.

(4A) Command 0117 (transmitter A to planar array antenna) only on each of four receiver-decoder combinations.

(4B) Point 2.

(4C) Point 2 repeated. Received *Lunar Orbiter* signal again for ≅18 min until *Lunar Orbiter* uplink was slewed.

(5) 3261 of point 23, then 3466 of point 23 (transmitter B to planar array antenna, wideband).

(6) 3364 of point 13 (transmitter B to planar array antenna, narrowband).

(7) 3261 of point 34, then 3365 of point 34 (transmitter A to omnidirectional antenna B, wideband).

(8) 3467 of point 24 (transmitter A to omnidirectional antenna B, narrowband).

(9) Point 35 (transmitter A to omnidirectional antenna A, narrowband).

(10) Point 57 (transmitter B to omnidirectional antenna A, narrowband).

(11) Point 66 (transmitter B to omnidirectional antenna A, narrowband, high power).

(12) Point 69 (transmitter A to planar array antenna, narrowband, high power).

(13) Point 70 (spacecraft shutdown).

For items (9) through (13) above, receiver search was confined to 2295 mHz ±30 kHz to complete this procedure prior to spacecraft sunset.

C. Spacecraft Engineering

1. Selected Midcourse Guidance Statistical Data—July 4, 1967 Direct Ascent Trajectories With 15- and 20-h Midcourse Maneuvers

A lunar miss figure-of-merit program has been executed using four covariance matrices that describe the injection statistics of four standard direct ascent trajectories, all in July 1967. These matrices include *Centaur* guidance platforms with Honeywell 177 accelerometers. The U and K matrices used were generated from precision trajectory data. Selected output data are presented in Table 7; Table 8 is a glossary of applicable terms.

The 3σ semimajor axes of the miss ellipses range in length from 1282 to 2275 km; 3σ unbraked impact velocity error varies from 5.67 to 9.06 m/s; and 3σ time-of-flight error ranges from 25.92 to 39.30 min—all evaluated for uncorrected miss. The sensitivities of terminal parameters to a velocity change at midcourse are given in Table 7 for the four trajectories for midcourse times of 15 and 20 h. Also presented are the 1σ velocities necessary to correct miss only, miss and velocity, and miss plus time of flight. The associated errors in time of flight and impact velocity are also tabulated.

The figures of merit, based on a worst-case trajectory, are tabulated for both miss only and miss plus time of flight. Accuracy of injection is deemed sufficient if the figure of merit (miss only) does not exceed 16.7 m/s. It should be noted that all figures of merit fulfill the accuracy requirements for all missions presented.

2. Mission D Spacecraft Orientation at Retroengine Ignition and Touchdown

Using the aiming points designated for Mission D and the July and August launch period trajectories (biased at midcourse), data have been generated to support premission analysis.

Figure 2 shows conventional angle definitions in spacecraft and in selenographic coordinates. Tables 9 and 10 show, as a function of launch day: aiming point; unbraked impact angle; the local vertical in spacecraft coordinates prior to retroengine ignition; DSIF in spacecraft coordinates at ignition and touchdown; sun in spacecraft coordinates at ignition and touchdown; touchdown roll orientation; and two preferred values of ϕ_{sun}.

Values of ϕ_E, ϕ_S (at ignition), β, and ρ in Tables 9 and 10 depend on the chosen values of ϕ_{sun} at touchdown.

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Table 7. Lunar miss figure-of-merit program data

July	Launch azimuth, deg	Declination of moon at arrival, deg	Earth-moon distance, km	Uncorrected miss					K matrix for midcourse = 15 h				K matrix for midcourse = 20 h			
				1σ sigma M1, km	1σ sigma M2, km	1σ theta, deg	1σ Delv, m/s	1σ Delft, min	$\overline{\lvert\nabla B \cdot T\rvert}$, km/m/s	$\overline{\lvert\nabla B \cdot R\rvert}$, km/m/s	$\overline{\lvert\nabla V\rvert}$, m/s m/s	$\overline{\lvert\nabla T_F\rvert}$, min/ m/s	$\overline{\lvert\nabla B \cdot T\rvert}$, km/m/s	$\overline{\lvert\nabla B \cdot R\rvert}$, km/m/s	$\overline{\lvert\nabla V\rvert}$, m/s m/s	$\overline{\lvert\nabla T_P\rvert}$, min m/s
15	89.46	−26.1346↑[a]	382,062	454.3	157.5	−43.24	1.89	8.64	182.6	170.7	0.57	2.34	163.0	155.4	0.528	2.010
15	115.0	−26.1346↑	382,062	758.4	147.4	−48.02	2.79	11.95	181.8	171.7	0.57	2.30	162.3	155.7	0.528	1.974
17	83.76	−18.8187↑	390,524	427.3	51.1	−21.94	1.90	9.07	187.6	165.9	0.57	2.41	166.6	153.5	0.525	2.062
17	105.95	−18.8187↑	390,524	626.3	62.5	−26.33	3.02	13.10	180.5	161.9	0.58	2.29	160.3	149.3	0.536	1.951

To correct miss only, midcourse = 15 h			To correct miss only, midcourse = 20 h			To correct miss and velocity, midcourse = 15 h		To correct miss and velocity, midcourse = 20 h		To correct miss and time of flight midcourse = 15 h		To correct miss and time of flight midcourse = 20 h		Worst case	
Velocity, m/s	Delv, m/s	Delft, min	Velocity, m/s	Delv, m/s	Delft, min	Velocity, m/s	TF error, min	Velocity, m/s	TF error, min	Velocity, m/s	V_{IMP} error, m/s	Velocity, m/s	V_{IMP} error, m/s	miss only, m/s	Miss and time of flight, m/s
2.576	1.345	7.253	2.906	1.462	7.735	3.702	1.352	4.139	1.971	4.141	0.306	4.886	0.496	6.348	7.853
4.089	1.804	9.372	4.644	2.023	10.279	5.430	1.555	6.176	2.414	5.914	0.358	7.055	0.620	9.604	11.896
2.260	1.442	7.894	2.557	1.541	8.298	3.620	1.443	4.024	2.078	4.105	0.320	4.838	0.510	6.327	7.809
3.434	2.220	11.050	3.888	2.406	11.791	5.445	1.774	6.116	2.742	6.082	0.424	7.266	0.727	9.960	12.343

[a]↑ Denotes ascending declination.

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Table 8. Glossary

Name	Definition
Sigma M1	1σ semimajor axis of uncorrected miss ellipse in B-space, km
Sigma M2	1σ semiminor axis of uncorrected miss ellipse in B-space, km
Theta	Orientation angle between $B \cdot T$ (earth equatorial plane) and semimajor axis, deg. Angle is measured from $B \cdot T$ positive clockwise (toward $B \cdot R$).
Delv	1σ error in unbraked impact velocity, m/s
Delft	1σ error in time of flight, min
$\overline{\lvert \nabla B \cdot T \rvert}$	Change in $B \cdot T$ related to perturbation in midcourse velocity, km/m/s $\overline{\lvert \nabla B \cdot T \rvert} = \left\lvert \frac{\partial \overline{B} \cdot \overline{T}}{\partial \dot{x}} i + \frac{\partial \overline{B} \cdot \overline{T}}{\partial \dot{y}} j + \frac{\partial \overline{B} \cdot \overline{T}}{\partial \dot{z}} k \right\rvert$
$\overline{\lvert \nabla B \cdot R \rvert}$	Change in $B \cdot R$ related to perturbation in midcourse velocity, km/m/s
$\overline{\lvert \nabla V \rvert}$	Change in unbraked impact velocity related to perturbation in midcourse velocity, m/s/m/s
$\overline{\lvert \nabla TL \rvert}$	Change in time of flight related to perturbation in midcourse velocity, min/m/s
Velocity to correct miss only	Equal to (Trace)$^{1/2}$ $\{K^{-1} U \Gamma U^T (K^{-1})^T\}$, where U and K matrices map injection errors to moon and from moon back to midcourse. Γ is injection covariance matrix. Velocity and time of flight terms are not included in U and K matrices.
Velocity to correct miss and velocity	Equal to (Trace)$^{1/2}$ $\{K^{-1} U \Gamma U^T (K^{-1})^T\}$, where U and K matrices include derivatives of impact velocity with respect to injection position and velocity and midcourse velocity.
Velocity to correct miss and time of flight	Equal to (Trace)$^{1/2}$ $\{K^{-1} U \Gamma U^T (K^{-1})^T\}$, where U and K matrices include derivatives of flight time with respect to injection position and velocity and midcourse velocity.
FOM (miss only)	Equal to (Trace)$^{1/2}$ $\{K^{-1} U \Gamma U^T (K^{-1})^T\}$, where U and K matrices represent worst-case trajectory. Velocity and time of flight terms are not included.
FOM (miss plus time of flight)	Equal to (Trace)$^{1/2}$ $\{K^{-1} U \Gamma U^T (K^{-1})^T\}$, where U and K matrices represent worst-case trajectory. Flight time derivatives are included.

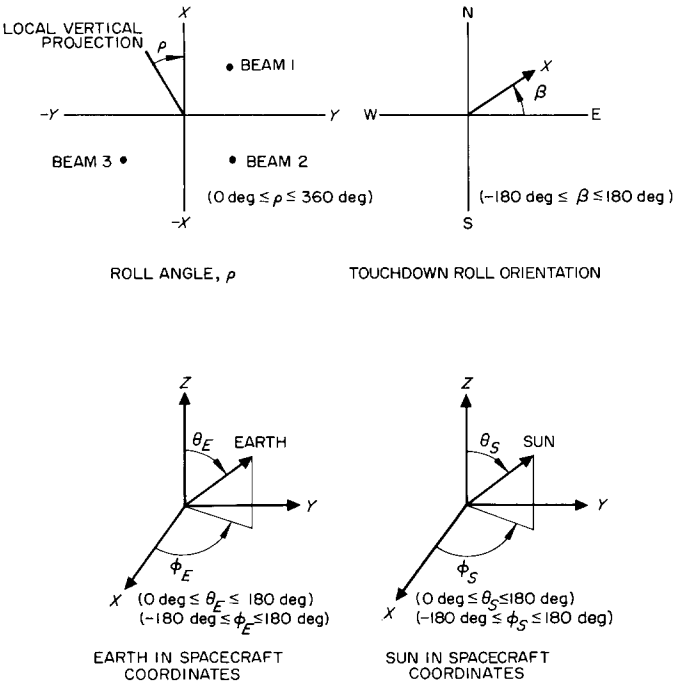

Fig. 2. Conventional angle definitions in spacecraft and in selenographic coordinates

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Table 9. Mission D data for July 1967

Parameters	Launch date, July									
	13		14		15		16		17	
Aiming point, deg	0.58N, 0.83W		0.58N, 0.83W		0.58N, 0.83W		0.58N, 0.83W		0.58N, 0.83W	
Incidence angle γ, deg	34.5		31.5		29.8		28.4		27.7	
Local vertical projection at retroengine ignition (ρ), deg	180.8	0.8	176.8	356.8	172.3	352.3	167.2	347.2	161.7	341.7
DSIF in spacecraft coordinates at ignition, deg										
θE	142.4	142.4	144.1	144.1	144.7	144.7	146.1	146.1	146.3	146.3
φE	178.4	−1.6	−179.9	0.1	−178.5	1.5	−176.4	3.6	−173.4	6.6
DSIF in spacecraft coordinates at touchdown, deg										
θE	177.2	177.2	175.5	175.5	173.9	173.9	172.6	172.6	171.5	171.5
φE	169.9	−10.1	159.8	−20.2	152.3	−27.7	149.2	−30.8	148.4	−31.6
Sun in spacecraft coordinates at ignition, deg										
θS	73.3	73.3	87.5	87.5	101.5	101.5	115.9	115.9	129.2	129.2
φS	−166.0	14.0	−167.3	12.7	−167.5	12.5	−166.5	13.5	−163.9	16.1
Sun in spacecraft coordinates at touchdown, deg										
θS	107.0	107.0	118.7	118.7	131.5	131.5	144.1	144.1	156.9	156.9
φS (required)	−166.0	14.0	−166.0	14.0	−166.0	14.0	−166.0	14.0	−166.0	14.0
Soft touchdown roll orientation β, deg	−164.6	15.4	−164.6	15.4	−164.5	15.5	−164.2	15.8	−163.4	16.6
Preference[a]	1	2	1	2	1	2	1	2	1	2

[a] 1 ϕ_{sun} (touchdown) $\triangleq$ −166 deg.
2 ϕ_{sun} (touchdown) $\triangleq$ +14 deg.

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Table 10. Mission D data for August 1967

Parameters	Launch date, August											
	10		11		12		13		14		15	
Aiming point, deg	0.58N, 0.83W		0.58N, 0.83W		0.58N, 0.83W		0.58N, 0.83W		0.58N, 0.83W		0.58N, 0.83W	
Incidence angle γ, deg	+31.2		+29.2		+27.6		+27		+26.5		+25.8	
Local vertical projection at retroengine ignition (ρ), deg	+178.3	+358.3	+173.6	+354.6	+168.7	+348.7	+163.6	+343.6	+159.6	+339.6	+156.9	+336.9
DSIF in spacecraft coordinates at ignition, deg												
θE	+143.5	+143.5	+144.6	+144.6	+146	+146	+146.4	+146.4	+146.8	+146.8	+147.9	+147.9
φE	+179.6	− 0.4	−179.0	+ 1.0	−177.3	+ 2.7	−175.1	+ 4.9	−173	+ 7	−171.3	+ 8.7
DSIF in spacecraft coordinates at touchdown, deg												
θE	+174.8	+174.8	+173.4	+173.4	+172.1	+172.1	+171.3	+171.3	+170.8	+170.8	+170.7	+170.7
φE	+168.5	− 11.5	+158.6	− 21.4	+153.5	− 26.5	+149.8	− 30.2	+148.4	− 31.6	+148.7	− 31.3
Sun in spacecraft coordinates at ignition, deg												
θS	+ 57.0	+ 57.0	+ 71.2	+ 71.2	+ 85.5	+ 85.5	+98.9	+ 98.9	+112	+112	+125.4	+125.4
φS	−163.6	+ 16.4	−165.7	+ 14.3	−166.2	+ 13.8	−165.6	+ 14.4	−164.2	+ 15.8	−162.3	+ 17.7
Sun in spacecraft coordinates at touchdown, deg												
θS	+ 87.5	+ 87.5	+100.4	+100.4	+113	+113	+126	+126	+138.7	+138.7	+151.1	+151.1
φS (required)	−166.0	− 14.0	−166.0	+ 14	−166	+ 14	−166	+ 14	−166	+ 14	−166.0	+ 14
Soft touchdown roll orientation β, deg	−164.6	+ 15.4	−164.6	+ 15.4	−164.6	+ 15.4	−164.6	+ 15.4	−164.5	+ 15.5	−164.2	+ 15.8
Preference[a]	1	2	1	2	1	2	1	2	1	2	1	2

[a] 1 ϕ_{sun} (touchdown) $\triangleq$ −166 deg.
2 ϕ_{sun} (touchdown) $\triangleq$ +14 deg.

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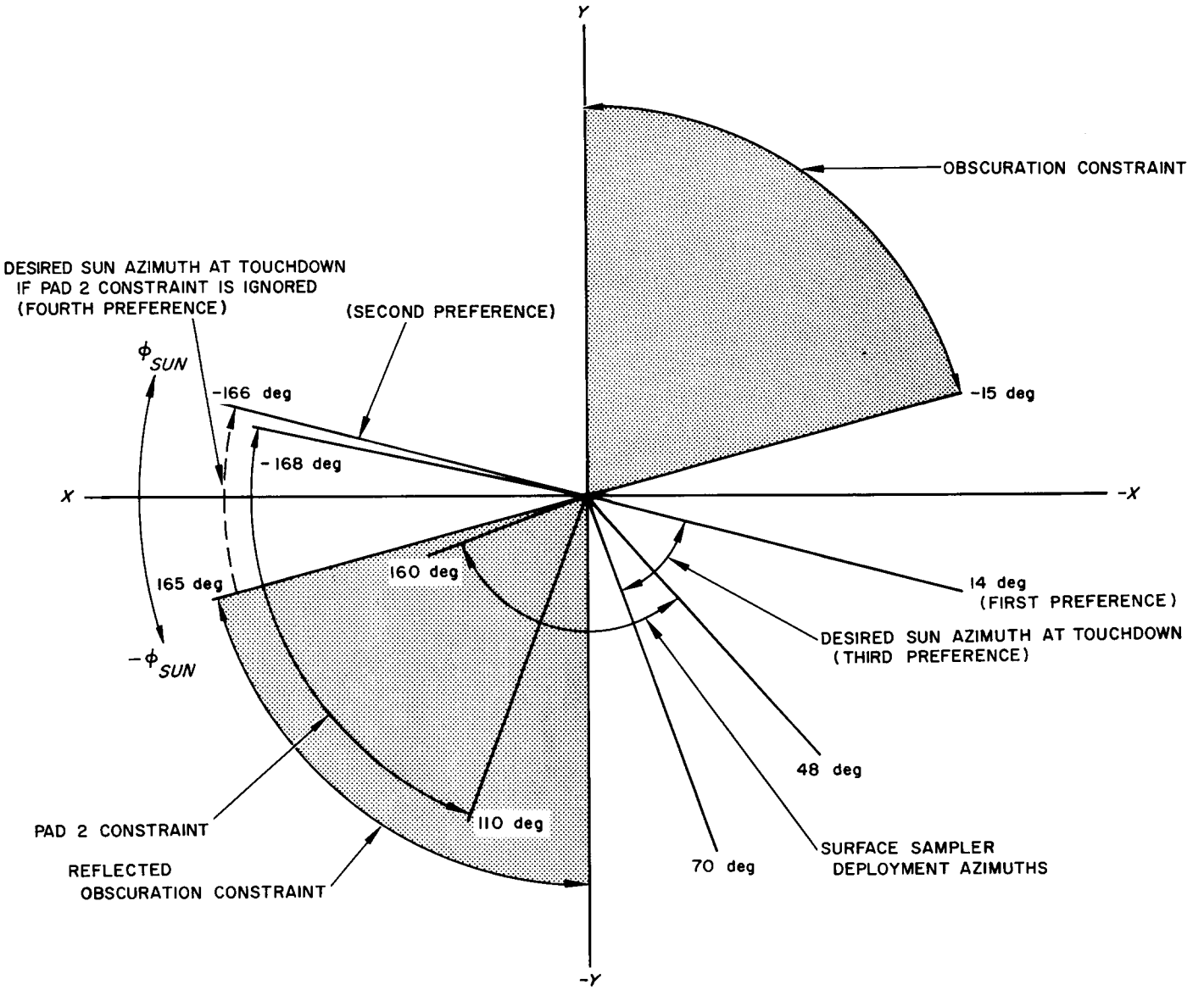

Fig. 3. Landed roll orientation

First and second preferences were chosen from Fig. 3. The angular conversion from Fig. 3 to the convention used in Fig. 2 has been carried through for the two chosen preferences.

3. Diversity Combining Enhancement of *Surveyor I* Touchdown Strain Gage Signals

Techniques can be applied to analog signals when such signals are received by two or more independent communication links. The *Surveyor I* touchdown strain gage link is a good example of space diversity. There are two reception paths, the 85-ft diameter *Pioneer* antenna and the 210-ft diameter Mars antenna. These provide two simultaneous analog signals with different noise characteristics and levels. Use of diversity combining techniques produces a better signal, i.e., lower noise level, than either of the received signals.

D. Structures, Mechanisms and Spacecraft Integration

1. Structural

a. Shock absorbers. Shock absorber serial number (SN) 11, assigned as an SC-4 spare, has been returned from the Air Force Eastern Test Range (AFETR) because of a scratch in the thermal paint. The unit is to be

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repainted and returned to AFETR in support of the SC-5 launch. Serial number 13, which was originally sent to AFETR to replace SN 11, failed the preload check at AFETR and has been returned to HAC.

b. Dynamic analysis.

Stability prediction by digital computer program. In an attempt to eliminate some of the conservatism in the *Surveyor* stability capability predictions, a revision has been made in the mathematical model. This change recognizes that the vehicle tends to topple about the edge of the downhill footpads rather than about the footpad hinge axis. As a result of the revision, the comparison between analytical and drop test stability boundaries was significantly improved.

Stability at low longitudinal impact velocities. Assessments of *Surveyor* spacecraft landing stability with low longitudinal and high lateral velocities have been made. Although this approach does not take into account vehicle shock absorber deflection during landing, it is felt that not accounting for deflection in this case is conservative.

The first case studied was that in which the spacecraft lands on the "double" feet and proceeds to topple about the double feet. This mode of instability is possible if the lateral velocity is sufficiently high. No instability occurred inside the 3σ touchdown dispersion ellipse, provided the longitudinal velocity is 2 fps or greater.

The second case studied concerned landing on the uphill single leg and then toppling about the downhill double legs.

Toppling about the double feet in the planar model is the most critical. However, no instabilities were found to occur for 3σ expected landing conditions, so that for the case of longitudinal velocities greater than 2 fps coupled with high lateral velocities, no instability is anticipated.

Vernier engine cutoff switch. As a backup mode to cutting off thrust power by radar control, touchdown cutoff switches are being considered. The feasibility of a vernier engine cutoff switch triggered by landing gear deflections has been investigated. A study was made of leg deflections for various landing conditions using the three-dimensional landing dynamics computer program which incorporates the flight control system. Figure 4 shows the vehicle configuration studied, and Table 11 lists the results for the various landing conditions. The investigation revealed that a longitudinal velocity greater than 3 fps is required for legs 2 and 3 contact on a 10-deg slope. Because of nonconservative aspects of the computer program (i.e., no consideration of spaceframe flexibility), a longitudinal velocity of 2 fps should probably be used as the minimum allowable to ensure switch activation.

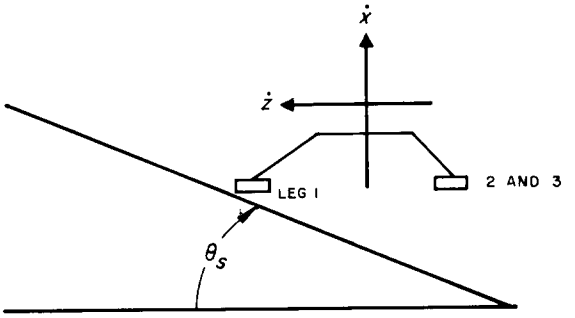

Fig. 4. Vehicle orientation

Table 11. Leg deflections for various landing conditions

θ_s, deg	$\dot{x}$, fps	$\dot{z}$, fps	Maximum leg deflections		
			Leg 1, deg	Leg 2, deg	Leg 3, deg
0	−1	0	0.61	0.61	0.61
0	−2	0	1.60	1.60	1.60
0	−3	0	3.63	3.63	3.63
2	−1	−3	0.31	0.41	0
2	−2	−3	1.11	3.10	2.44
2	−3	−3	2.14	5.77	5.77
10	−1	−5	0.12	0	0
10	−2	−5	0.50	0	0
10	−3	−5	1.46	0	0

c. Postmission analysis. A flight control system has been incorporated in the three-dimensional landing dynamics program in an attempt to duplicate the landing gear strain gage force-time histories for the first two *Surveyor III* landings. The three-dimensional soft surface program has been used to match the third touchdown, which occurred after the vernier engines were shut off. The best comparisons to date for the three events are shown in Fig. 5. Better comparisons are expected as refinements are made in the mathematical models.

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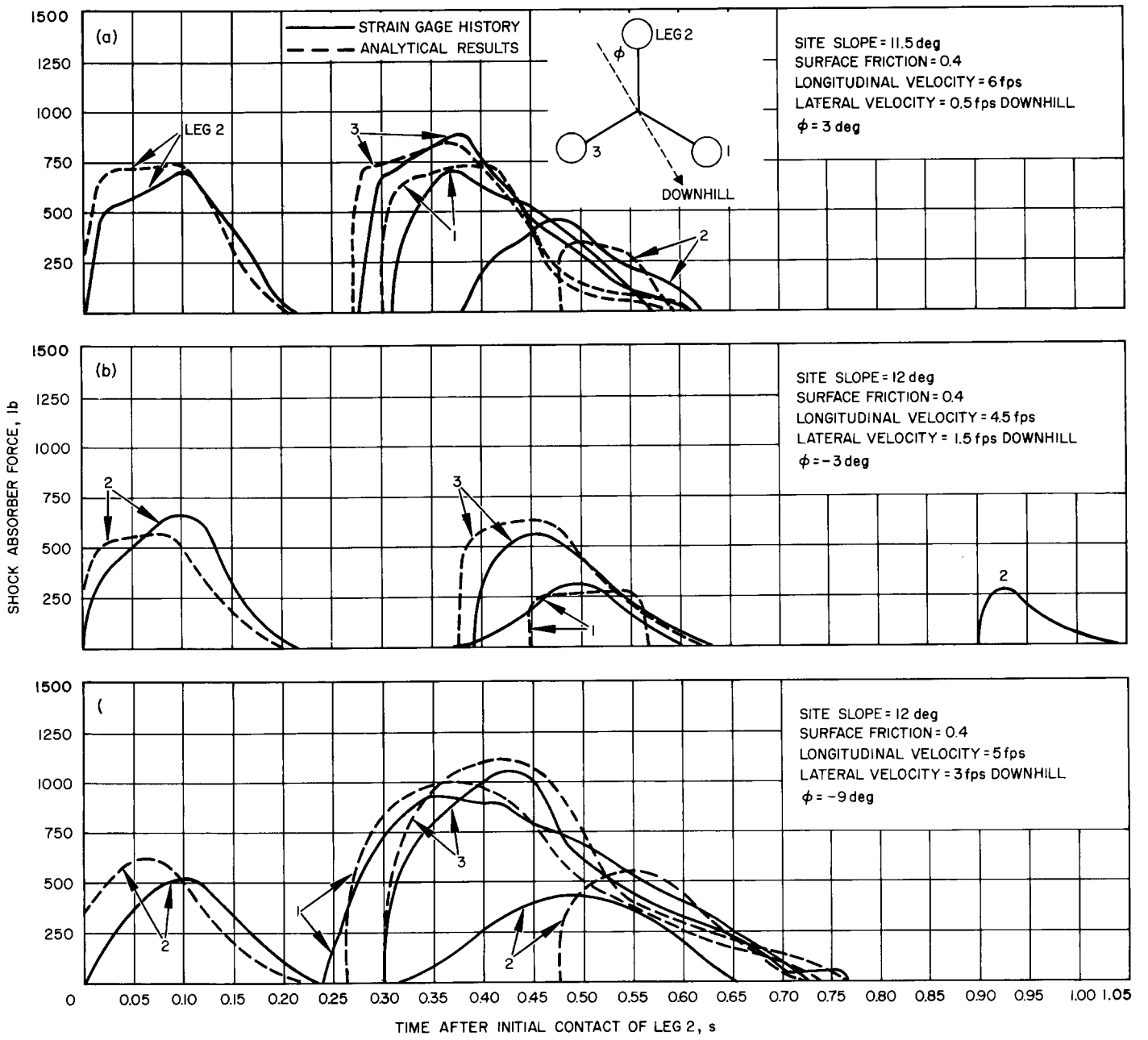

Fig. 5. Comparison of strain gage histories and analytical results for *Surveyor III* shock absorbers: (a) event 1; (b) event 2; (c) event 3

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d. Dynamic tests. The S-9 test model has been modified structurally to represent an SC-5 type spacecraft. The present S-9 vehicle configuration includes A/SPP 3035092 with a JPL-furnished solar panel, a flight-type alpha scattering mechanism with a dynamically simulated sensor head, and SC-5 type thermal compartments A, B, and C. Auxiliary TV viewing mirrors and an SC-5 type vernier engine 1 bracket have also been installed on the vehicle.

S-9 was instrumented with strain gages and accelerometers in preparation for the vibration tests which were completed June 30, 1967. Drop testing of the vehicle is scheduled to start July 11, 1967.

e. Structural tests.

Escapement hook test — alpha scattering experiment. The escapement hook, for quick attachment and removal of the sensor head from the alpha scattering escapement mechanism, has undergone verification tests at the structures test laboratory (Figs. 6 and 7).

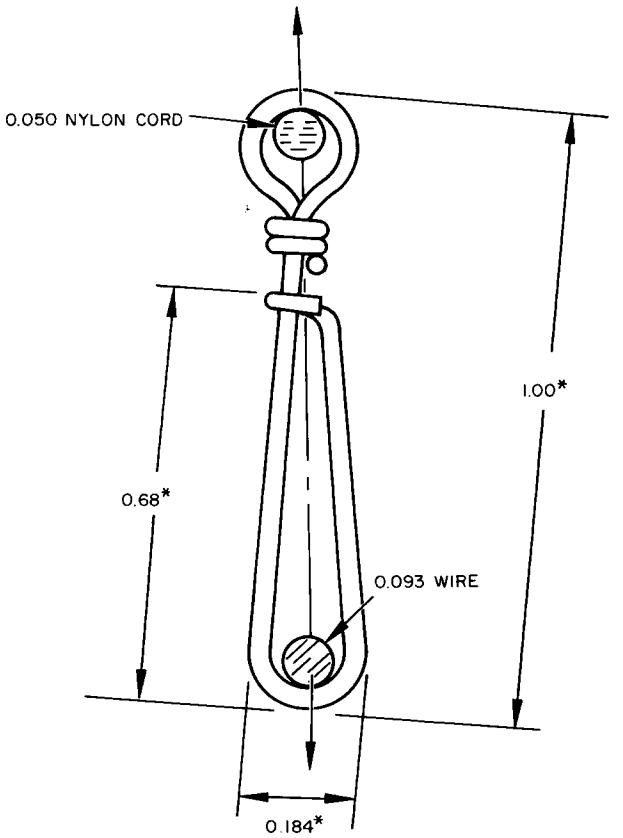
Three representative production hooks were tested simulating the actual loading conditions. All three hooks withstood more than 170% of the limit load and showed a negligible amount of permanent set.

Static calibration — upper compartment supports. In order to evaluate the effects on the upper fiberglass support struts of various combined loads applied to compartments A and B, six sample struts were instrumented with a system of strain gages to indicate strains associated with both axial and flexural stresses. The struts are plastic tubes consisting of a glass fabric base impregnated with epoxy resin.

Bending moments were applied in increments to a 250-lb maximum with strain readings recorded at each load.

For tensile and compressive tests, specimens were attached to a tensile machine and a tensile load was applied in increments up to a 1000-lb maximum, and the axial strain was recorded at each increment. Compressive load was similarly applied to a 500-lb maximum.

The mean value of the flexural modulus of elasticity was 3.53×10^6 psi; the mean value of the modulus of elasticity in tension is 3.303×10^6 psi; the mean value in compression is 3.302×10^6 psi.



* CONTROLLING DIMENSIONS

DIMENSIONS IN INCHES

Fig. 6. Alpha scattering experiment background count equipment

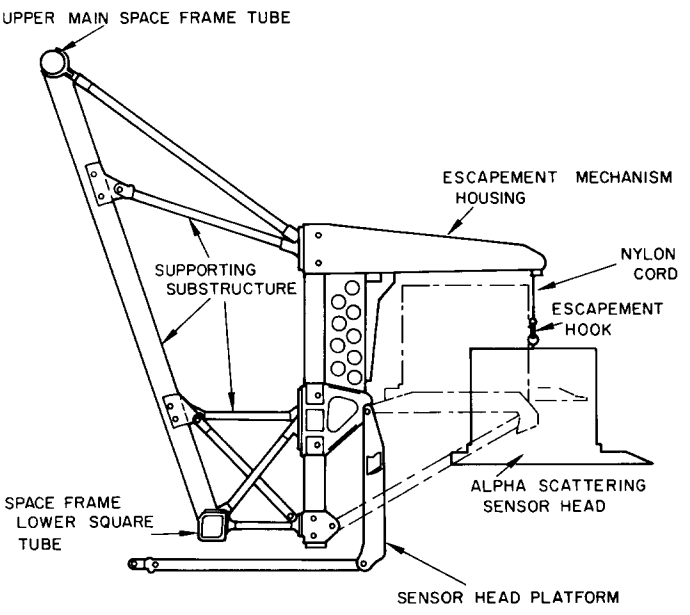


Fig. 7. Cord escapement hook

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The above results indicate that the values of the modulus of elasticity for the thermal compartment struts are comparable to those specified in Mil-Hdbk-17 for epoxy resin glass-fiber reinforcement using 181 fabric.

2. Mechanical

a. Antenna solar panel positioner. The A/SPP provides mounting, positioning, position indicators, and launch and landing locks for the planar array antenna and solar panel. The antenna support structure includes controllable axes for roll, elevation, solar, and polar rotations. The drive for each axis uses a solenoid stepper motor for power and a wirewound potentiometer for position indication.

A/SPP 3035092, SN 1. This unit, which incorporates the latest model high-torque motors, was originally assembled for use on SC-5. It completed its flight acceptance tests (FAT) successfully, and was installed on SC-5 for spacecraft system tests. During assembly of SN 2, however, a drive connector soldering problem was noted and corrected (see below). Serial number 2, therefore, replaced SN 1 on SC-5; the SN 1 drive connectors were reworked by bonding on a retaining doubler, and the unit was assigned to the A/SPP type approval test (TAT) program and will be installed on the S-9 vehicle. This program consists of a previbration alignment, vibration, drop program, postdrop alignment, thermal vacuum tests, and posttest inspection. The unit has completed previbration alignment, and is being instrumented for the drop program.

A/SPP 3035092, SN 2. As noted above, a drive connector solder joint became loose during drive FAT. The joint in question is the soldered connection of the connector shell to the drive housing. Since the failure was traced to poor workmanship and design, an engineering change was made incorporating riveted connector retaining doublers. This adds a mechanical constraint to the soldered joint.

Serial number 2 passed its FAT with no trouble failure reports (TFRs), and is now installed on SC-5.

A/SPP 3035092, SN 3. This unit was assembled for SC-6 using the 293275, serial number 1, structure. The drives were reworked to incorporate the latest model high-torque motors.

During production testing of motors for this unit, a snap ring problem was encountered. Motor stall instances were noted and, upon disassembly, it was found that several armature cam follower retaining rings had become disengaged, causing the motor to jam.

Further investigation revealed two discrepancies that seemed to be associated with the snap ring failures. It was noted that the cam follower bearings had not been burnished and ultrasonically cleaned after application of dry lubrication, resulting in excessive dry lube. Also, the cam follower inside diameters were undersize. These two discrepancies, in combination, resulted in a very rough bearing and the likelihood of inner race rotation on the cam follower pins. It was thought that this rotation opened the snap rings, permitting displacement from their pin grooves. The undersize inside bearing diameters had occurred in SN 1 and SN 2 A/SPP but were not recorded in their unit history records. The lack of inspection on these bearings allowed them to be installed in the S-3 A/SPP motors.

Special engineering tests discredited this theory by showing that deliberately frozen bearings would not necessarily produce the failure. In fact, the excessive dry lube was quickly burnished out, resulting in relatively free-running bearings.

Further engineering investigation revealed by demonstration that the snap rings could be easily misassembled, and that the error is not readily detected without close inspection. During the demonstrations, snap rings were deliberately pressed over the pin next to the groove but not engaged in the groove. The snap rings yielded in the process but remained firmly on the shaft through friction alone. The extra force required to make this assembly error was not easily discernible.

Continued investigation revealed that all these motor sets of snap rings had been installed by a workman who lacked previous experience in this operation. Also, this operation had not been accomplished or inspected under magnification. This procedure has since been changed.

Serial number 3 was assembled and put into FAT. During thermal vacuum tests at low temperature, long flats appeared on the solar axis potentiometer records, indicating the potentiometer was not rotating even though the solar drive was stepping properly. Thermal vacuum testing was interrupted, the solar drive replaced, and thermal vacuum testing resumed. A similar occurrence developed again on the solar axis, and at low temperature. Flight acceptance testing was completed and, while stepping the roll axis to the transit position in accordance

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with the test procedure, a flat appeared on the roll potentiometer record.

Investigation revealed that the potentiometer wiper is constrained rotationally on a KEL-F bushing only by the friction produced by a press fit. It is believed that the KEL-F flows and loses some of the press at high temperature and then, because of differential contraction, becomes loose at low temperature. This was substantiated by inspection of a used potentiometer in which a rotationally loose wiper was found. The potentiometer vendor confirmed the possibility of such a failure mode by cold soaking a wiper and shaft assembly and noting the resulting wiper looseness.

Potentiometers are being reworked by the vendor. The bushing material is being changed to an epoxy-filled, laminated fiberglass; the shaft is being knurled; and the wiper is being pinned to the bushing.

b. High-torque stepper motor and transmission. A lot of five pin and bushing high-torque stepper motors was taken from initial production runs and delivered to engineering for design verification and qualification tests.

Three of the motors were used for design verification tests. The electrical characteristics of the motor coils were first checked, and then the motors were tested for stall torque at ambient temperature of $-225°F$. The mating transmissions were checked for efficiency, the drives assembled, and the drive stall torque measured.

In each design verification test, the drive was operated with a sinusoidally varying load of ± 155 in.-lb for 65,000 steps, reversing direction of rotation at 15-min intervals. Stall torque measurements were taken at 20,000-step intervals. The minimum current to step with maximum opposing and aiding loads for both directions of rotation was measured at 5000-step intervals. The output shaft load was then reduced to ± 100 in.-lb, and ambient temperature reduced to $-225°F$. Stepping was continued for 40,000 steps, with a minimum current measurement taken at 2500-step intervals. The temperature was returned to ambient and, following a stall torque measurement, stepping was continued with minimum current tests at 5000-step intervals. Stall torque measurements were taken at 20,000-step intervals until the torque dropped to 160 in.-lb with a 10.0-amp pulse current amplitude.

Motor SN 2 failed after 213,521 steps; SN 4 after 213,982 steps; and SN 6 after 270,406 steps. The required motor life is 120,000 steps for the roll, solar, and elevation axes, and 72,000 steps for the polar axis.

Two production motors were assembled into drives by Manufacturing for the qualification test program. The drives were subjected to production flight acceptance tests prior to delivery to Engineering. A polar axis drive and an elevation axis drive were subjected to qualification tests.

Both drives were tested for electrical characteristics and then given a functional performance test to determine stall torque at specified pulse current amplitudes. Each drive was subjected to all significant prelaunch, launch, and lunar operational environmental conditions, including shock, vibration, and thermal vacuum of 1×10^{-5} torr at temperatures in the range of -225 to $+250°F$. Stall torque measurements were repeated at ambient conditions after each exposure to determine the capability of each drive to operate with 200% of lunar load requirements with specified minimum pulse current amplitudes.

The polar drive was subjected to a minimum of 33,400 steps at ambient temperature and pressure; 10,000 steps at ambient temperature and at vacuum; 10,000 steps at low temperature and at vacuum; and 18,600 steps at high temperature and at vacuum.

The elevation drive was subjected a minimum of 61,400 steps at ambient temperature and pressure; 15,000 steps at ambient temperature and at vacuum; 15,000 steps at low temperature and at vacuum; and 28,600 steps at high temperature and at vacuum.

Upon completion of the engineering qualification tests, both drives were subjected to life tests until failure, as defined by a stall torque output of less than 9 in.-lb for the polar drive and 102 in.-lb for the elevation drive, and specified minimum currents.

Each drive was first tested to determine the stall torque capacity and to determine the capability of meeting the *Surveyor* equipment specification, which requires a torque capacity of 150% of lunar load at the end of design life at -225 and $+250°F$. At minimum current, the polar drive capacity was 290% for both high and low temperatures. The elevation drive capacity was 300% at low temperature and 380% at high temperature.

The polar and elevation drives were then operated continuously with sinusoidally varying loads under ambient

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conditions to determine their ultimate lives. Stall torque measurements were made at 20,000-step intervals. Steps were distributed equally between clockwise and counterclockwise directions of rotation.

The polar drive stall torque dropped below 9 in.-lb after 200,562 steps. The elevation drive operated for 338,665 steps before the stall torque dropped below 120 in.-lb.

Qualification test and design verification test data are presented in Figs. 8 and 9.

c. Thermal switches. Active elements of the *Surveyor* compartment thermal control are the nine thermal switches on compartment A and the six thermal switches on compartment B. These bimetallically-operated units either make or break a conductive path to fixed radiators as a function of compartment temperatures to maintain the temperature within a narrower band than the external temperature excursions.

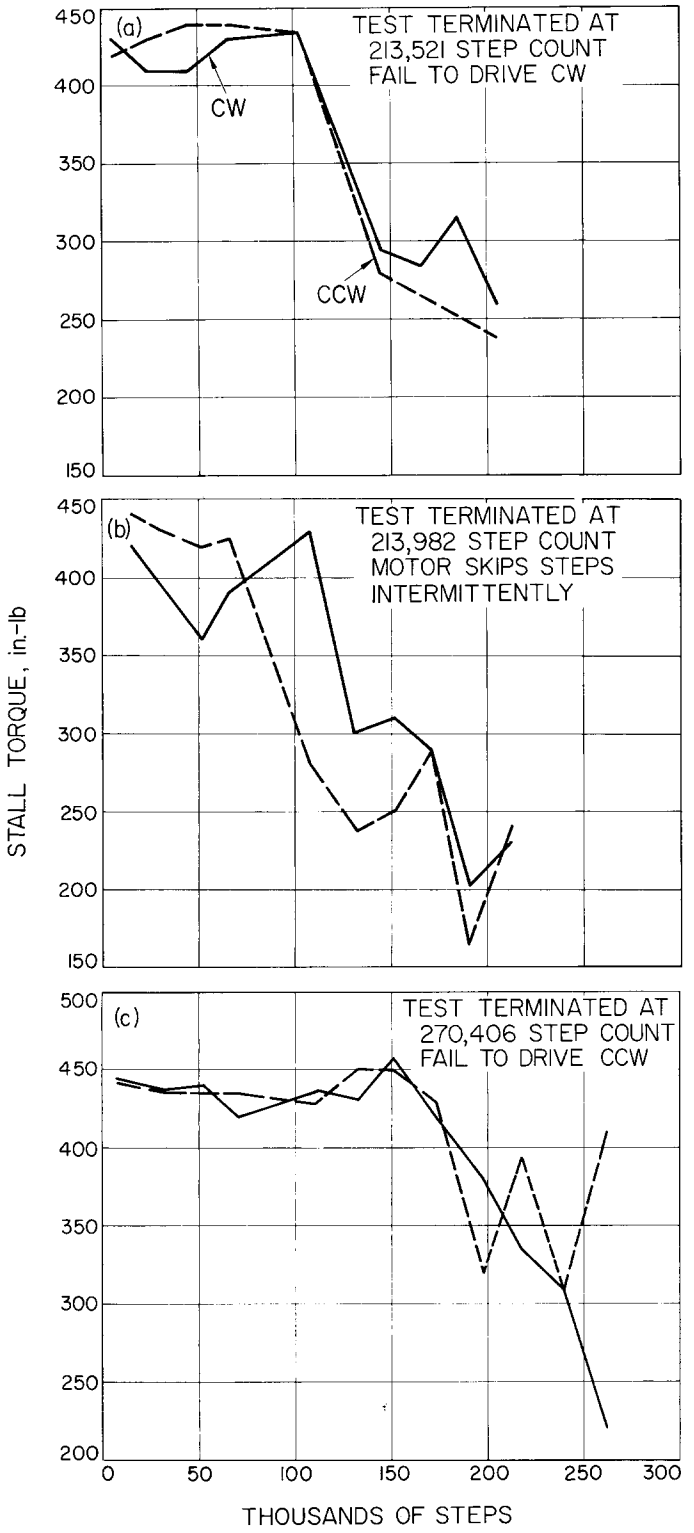

Fig. 8. High-torque stepper motor design verification test: (a) SN 2; (b) SN 4; (c) SN 6

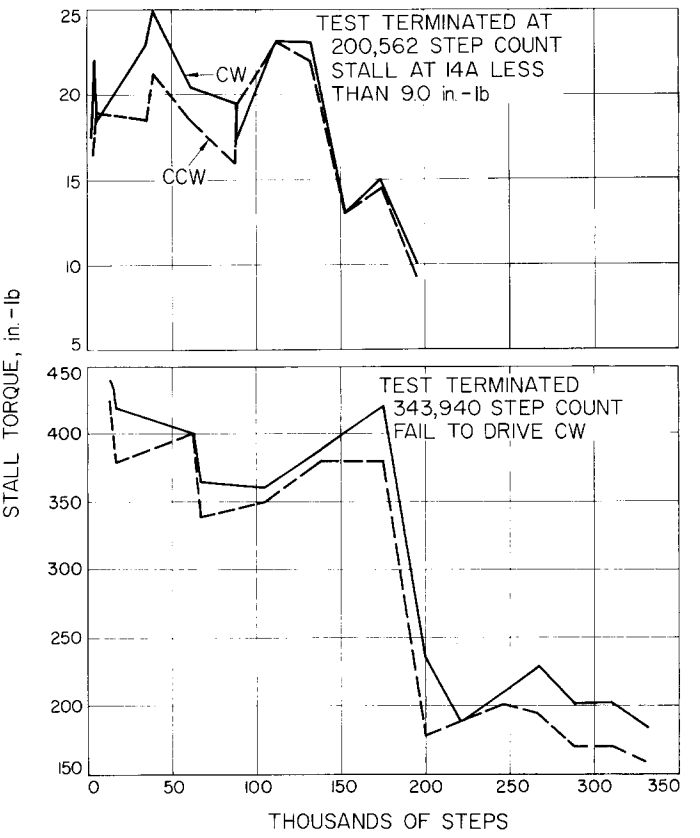

Fig. 9. High-torque stepper motor engineering qualification test, SN 1: (a) polar drive; (b) elevation drive

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d. Omnidirectional antenna mechanisms. Omnidirectional antenna mechanisms A and B use tapered fiberglass cloth tubes bonded to aluminum hinge bracket assemblies. They deploy and lock the antennas into operating position before the spacecraft separates from the *Centaur*.

e. Separation sensing and arming device. Three separation sensing and arming devices for each spacecraft provide redundant activation of the postseparation squib firing circuits and vernier engine solenoid circuit after the spacecraft separates from the *Centaur*.

f. Auxiliary TV viewing mirrors. For SC-5 and -6, spherical convex beryllium base mirrors replace the two plane mirrors used on SC-3 and -4. These mirrors view the areas below the vernier engines and crushable blocks in the vicinity of legs 2 and 3.

g. Pyrotechnic devices. Engineering liaison was provided at AFETR prior to J-FACT and launch of SC-4 for checkout and installation of all spacecraft pyrotechnics.

Pin pullers. To ensure full stroke at conditions of full load and minimum temperature, the A/SPP pin pullers were modified to include vented end plugs. This change was incorporated into SC-3 and all subsequent spacecraft.

Testing of deliberately misaligned pin puller shafts was completed without reaching a conclusion as to the cause of the shaft breakage encountered during A/SPP design verification testing. The testing failed to reproduce the failure even under a combined condition of excessively low temperature and extreme misalignment.

The anomalous pressure readings previously reported were eliminated by the replacement of the change amplifier and oscilloscope. Figure 10 shows simultaneous records of ballistic pressure and shaft tension at low ambient temperatures taken with the new instrumentation.

3. Integration

Simulated *Atlas/Centaur* launch profile tests have been performed on the SC-5 type A, B, and C compartments to determine the susceptibility of the thermal blanket to rupture during rapid evacuation of the compartments during launch. Posttest disassembly of the compartments disclosed no blanket damage.

a. Flight and test vehicle status.

SC-4 flight spacecraft. SC-4 is at AFETR, and preflight preparations are proceeding on schedule.

SC-5 flight spacecraft. Vernier engine vibration has been successfully completed and the spacecraft is being prepared for shipment to General Dynamics/Convair June 17 for combined systems test.

Supplementary ballast, required to help maintain SC-5's center of gravity within required limits, was installed in sector 2.

After solar thermal vacuum testing, it was necessary to replace the test thermal blankets in the compartments with flight-type blankets. This operation requires that the mirrors be removed from the thermal switches, and is normally accomplished between phase B and C of solar

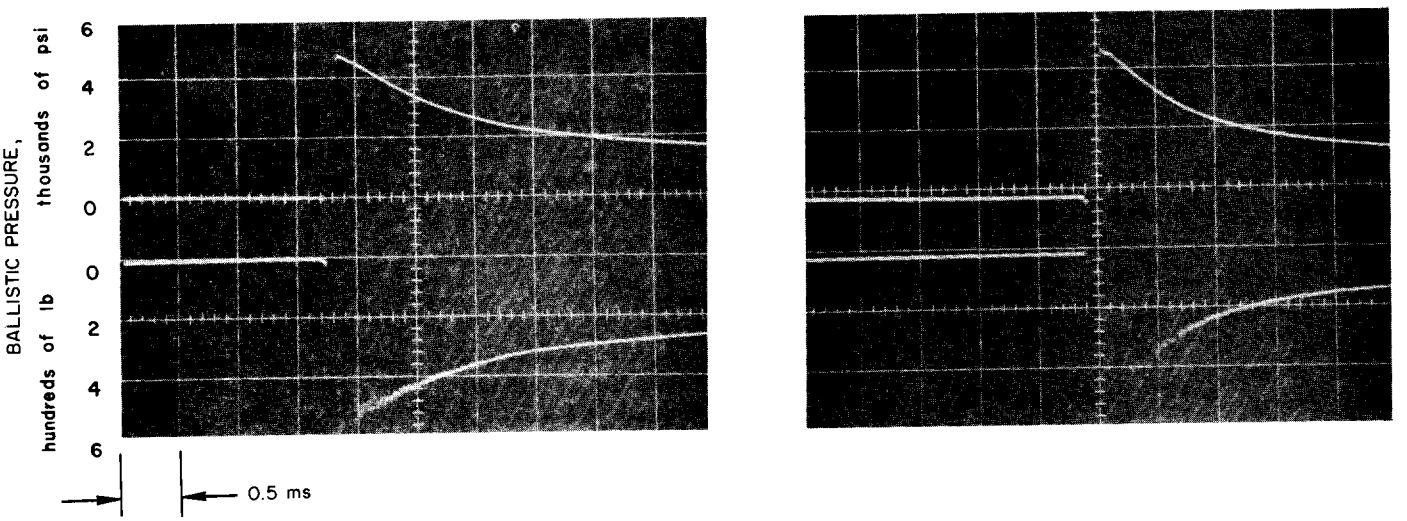

Fig. 10. Ballastic pressure and shaft tension records: (a) ambient temperature; (b) low (− 140°F) temperature

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thermal vacuum. Switch performance in phase C then verifies their proper functioning. For SC-5, however, the phase-C requirement was waived, and a thorough cleanliness inspection was performed to check out the switches. They were acceptable, with the exception of one switch on the B compartments which has been replaced.

SC-6 flight spacecraft. SC-6 has completed initial system sine checkout, and preparations for mission sequence/electromagnetic interference and solar thermal vacuum testing are under way.

Leg 1 failed the deployment test during flight approval testing. Pieces of Teflon were found inside the lock strut and at one of the bearing surfaces. An oversize screw and an undersized hole were found, and there is evidence that the leg journal shaft was out of round. The leg has been reworked and now deploys within specification limits. SC-5 and SC-7 will be inspected for similar problems. An engineering change request has been issued to remedy the out-of-round condition on the landing gear shaft.

SC-7 flight spacecraft. Modifications for the installation of the SM/SS on SC-7 have been completed and initial system checkout has resumed.

S-10 thermal test model. S-10 is being upgraded to A-21E configuration.

E. Power Systems

1. Solar Panel

Solar panel serial number (SN) 5 was processed through an electrical sunlight test at Table Mountain and shipped to Air Force Eastern Test Range (AFETR) May 9. An acceptance test was performed May 27, and the panel was placed in bonded stores to await installation on SC-4 June 12. Serial number 4 was returned to HAC from AFETR for processing through an electrical sunlight test at Table Mountain. The test was conducted May 16 and the panel was shipped back to AFETR May 20. On May 25, an acceptance test was completed and the panel was placed in bonded stores as a spare unit.

Solar panels SN 4 and 5 (part No. 251167 for SC-5 and up) were subjected to approximate type approval test level sine wave inputs in three axes during the conduct of flight acceptance vibration testing. Sine levels were exceeded only in the frequency range of 100 to 1500 Hz. A joint HAC-JPL meeting was held May 16 to analyze data and to determine the flight quality of the panels. Based on the following considerations, it was the conclusion of this meeting that the panels had not been overstressed and were acceptable for flight.

(1) The overvibration (higher g peak levels than specified) occurred in the high frequency range above the first resonance, 60 to 65 Hz, and the second resonance, 90 to 95 Hz of the panel. The stresses induced on the panels in that frequency range were of low magnitude and no significant damage had resulted.

(2) No damage to the electronics had been experienced on any of the panels after vibration, flight acceptance, and type approval test levels.

(3) The flight panels are expected to be exposed to only one cycle of spacecraft vibration level.

(4) The design data show a satisfactory margin.

However, to be conservative, solar panel SN 5, which had been tested to slightly higher levels than SN 4 on the basis of integrated inputs, was designated the spare unit, with SN 4 allocated to SC-7, SN 2 to SC-6, and SN 3 to SC-5.

2. Batteries

a. Main battery

Discharge data. The A-21E lunar simulation test operated seven batteries at various discharge rates and temperatures over the range of 30 to 110°F. The results of this testing have been used to update the average voltage versus temperature plot (Fig. 11).

Also, to gain low temperature operating information on an almost fully discharged battery (end of lunar night conditions), SN 99 was operated to simulate charging and discharging at temperatures from 0 to −40°F. Figure 12 illustrates the results of charging the main battery at 0 to −40°F at 0.5 to 3 A, with Fig. 12(a) showing initial voltage and Fig. 12(b) showing voltage after 1 min of charge. From the figures it can be seen that battery terminal voltage varies directly with charge current and inversely with temperature.

Figure 13 presents discharge data for the main battery at 0 to −40°F and rates of 0.5 to 12 A. Figure 13(a) shows initial voltage, and Fig. 13(b) shows voltage after 1 min of discharge. In addition, pulse discharge at simulated TV and antenna solar panel positioner (A/SPP) loads is presented. The figures show that discharge voltage varies directly with current and inversely with temperature and that pulse voltages are slightly higher than steady-state voltages.

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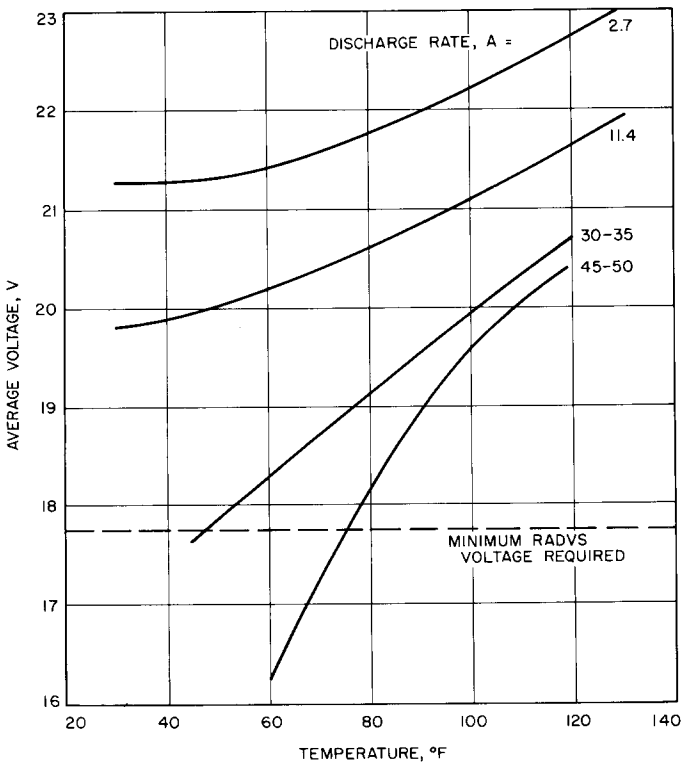

Fig. 11. Main battery average voltage vs temperature at various operating rates

Mission sequence tests. The purpose of these tests was to establish the A-21E *Surveyor* main battery parameters during a typical SC-5, -6, or -7 mission sequence. Specifically, the test was designed to demonstrate satisfactory spacecraft operation at temperatures ranging from 57 to 95°F. The test sequence simulated transit, landing, and lunar operations.

During the report period, testing was concluded and a final report was prepared and distributed. Seven batteries (six complete missions and one partial mission) were successfully operated through the simulated mission sequence. The tests demonstrated that the *Surveyor* main battery can:

(a) Provide terminal descent voltages in excess of 18.0 V at temperatures greater than 80°F (launch, transit, and touchdown test phase)

(b) Operate over a controlled temperature exposure range of 30 to 125°F for 1 lunar day.

(c) Provide an estimated 176 A-h of capacity for lunar night survival, based on the completion of six tests, each of 9 days duration.

Figures containing anticipated battery voltages on charge and at TV and A/SPP discharge rates over the temperature range of 30 to 125°F have been prepared to provide a guide for estimating battery voltage under typical operating conditions (Fig. 14). An estimate of the time required to fully recharge a main battery, assuming a typical range of available capacities at touchdown and lunar operations, has been prepared to help predict battery energy available during lunar operations (Fig. 15).

SC-3 flight. Serial number 108 main battery was installed on SC-3, and its performance was satisfactory. Figure 16 was plotted from information received in real time and compared to the SC-3 solar thermal vacuum (STV) test information to determine battery condition as the mission progressed. Information after 63 h was unavailable because of telemetry difficulties. A similar plot will be made for the SC-4 mission.

SC-3 reliability test. Serial number 106 main battery was designated as a reliability test battery for the SC-3 mission. The battery proceeded through a normal flight acceptance test sequence and was then subjected to an A-21E lunar surface operation simulation sequence at a nominal temperature profile as follows:

Day	Temperature, °F
1	82
2	77
3	83
4	88
5	93
6	98
7	63
8	43
9	30

Following the lunar surface operation the battery was discharged at a 7.0-A rate while the temperature was lowered approximately 4°F/h. At 23°F, one cell reversed, and the discharge rate was lowered to 200 mA. The battery was stabilized at −40°F while TV, A/SPP, and charging (using a solar panel simulator) operations were simulated. The battery was returned to room temperature at approximately 6°F/h and discharged at a 5-A rate. Successful completion of the reliability test was signified by the battery's ability to accept a full recharge at 2 A in a normal manner.

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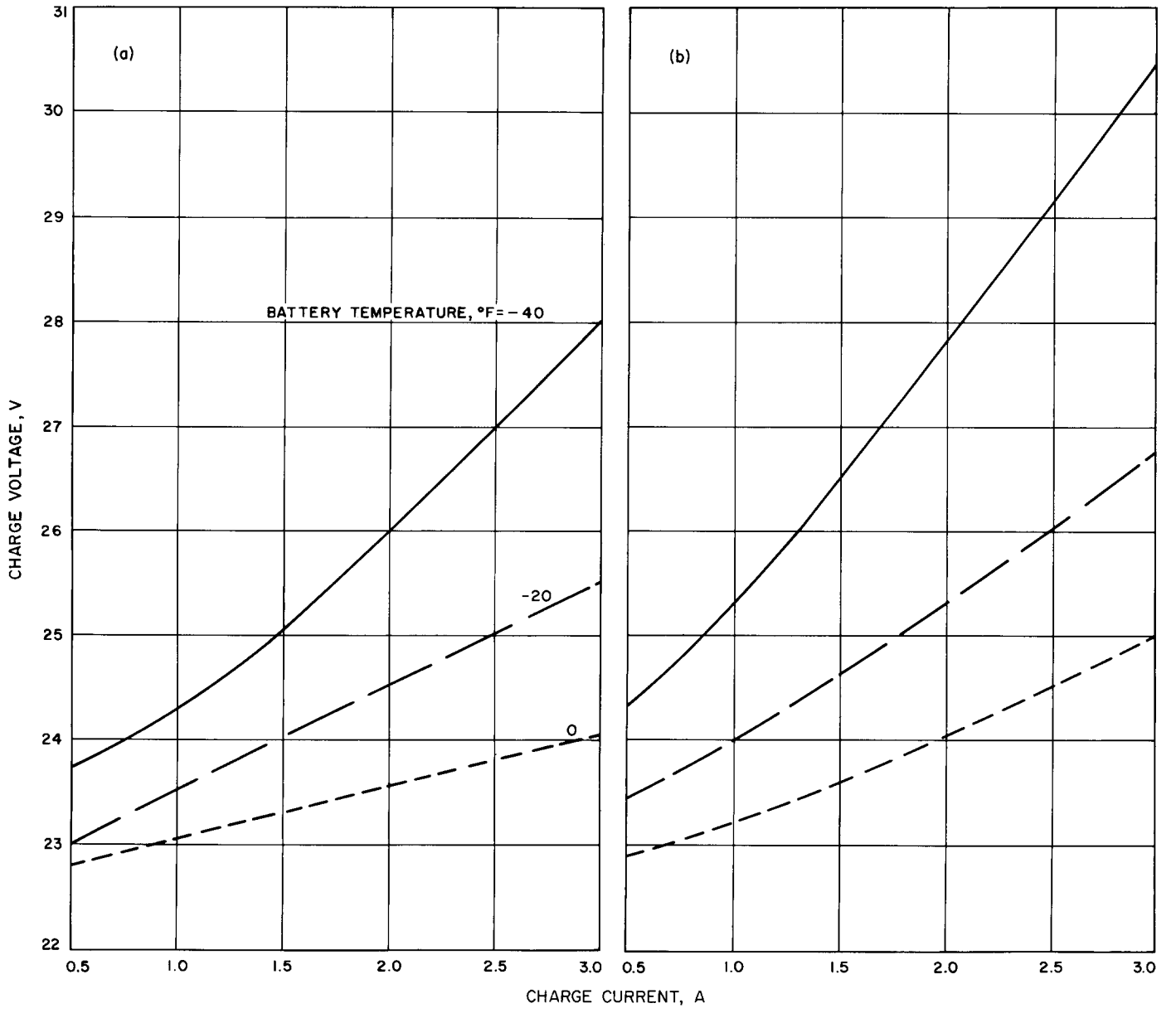

Fig. 12. Charge voltage vs charge current at low temperatures: (a) initial charge voltage; (b) charge voltage after 60 s

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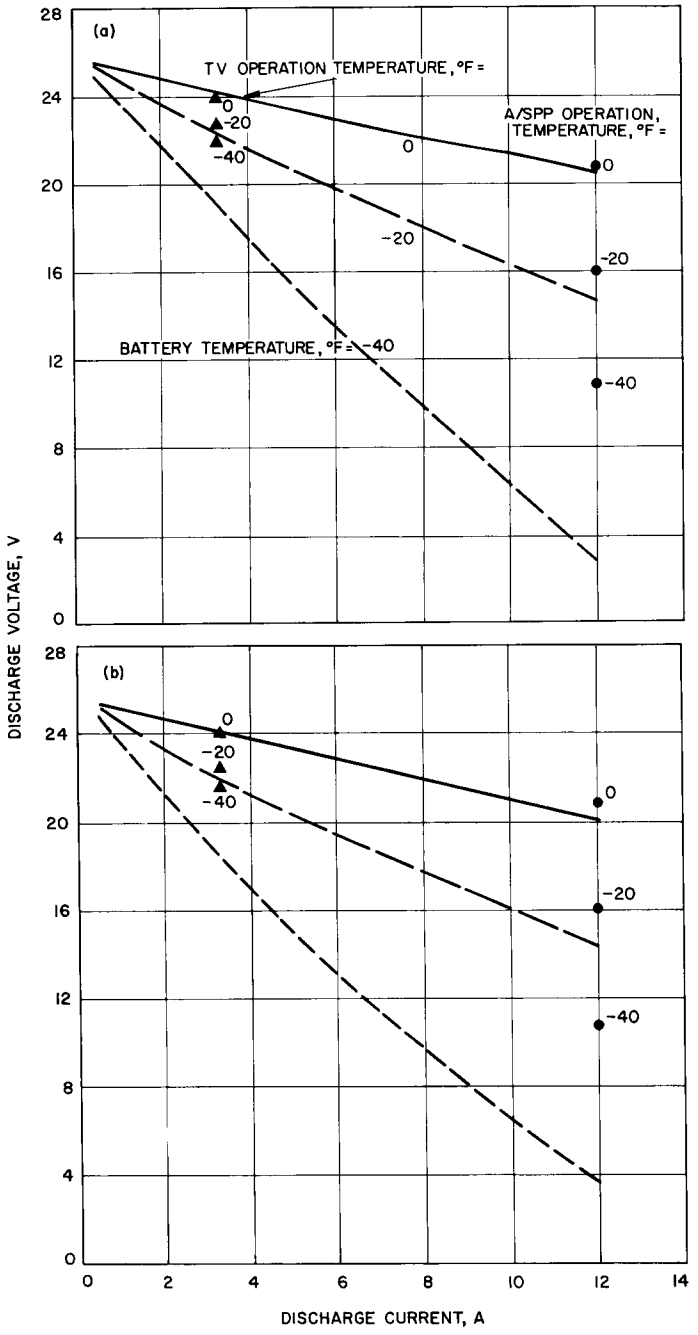

Fig. 13. Discharge voltage vs discharge current at low temperatures: (a) initial discharge voltage; (b) discharge voltage after 60 s

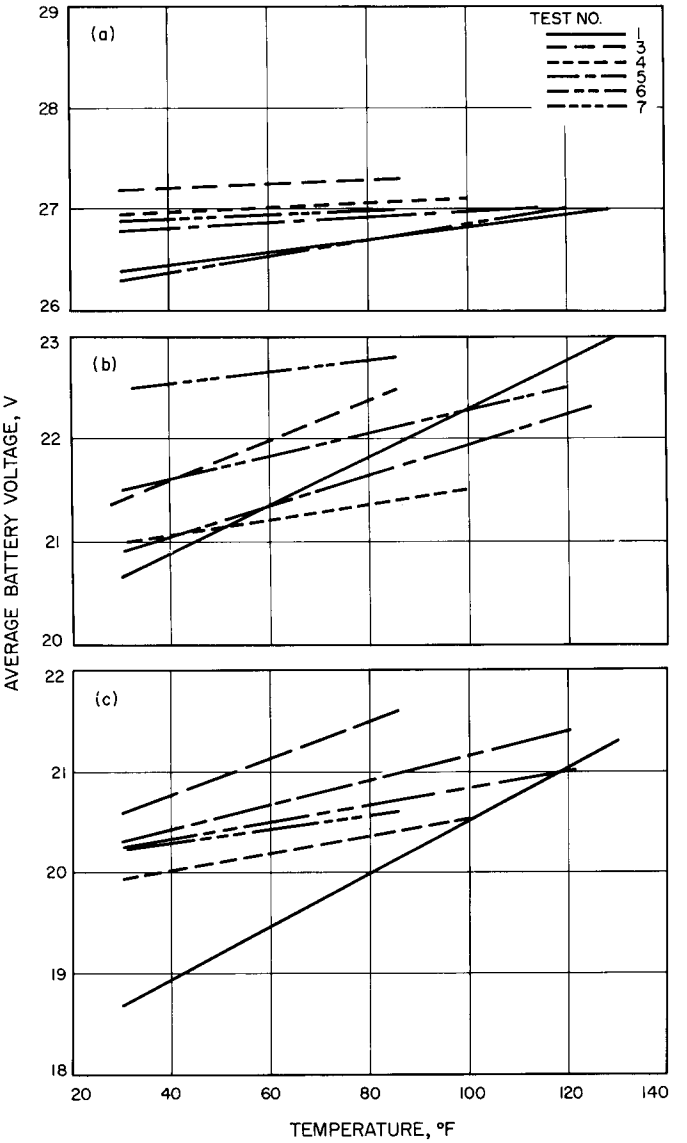

Fig. 14. A-21E main battery lunar-operating-tests average battery voltage vs temperature: (a) charge; (b) TV discharge; (c) A/SPP discharge

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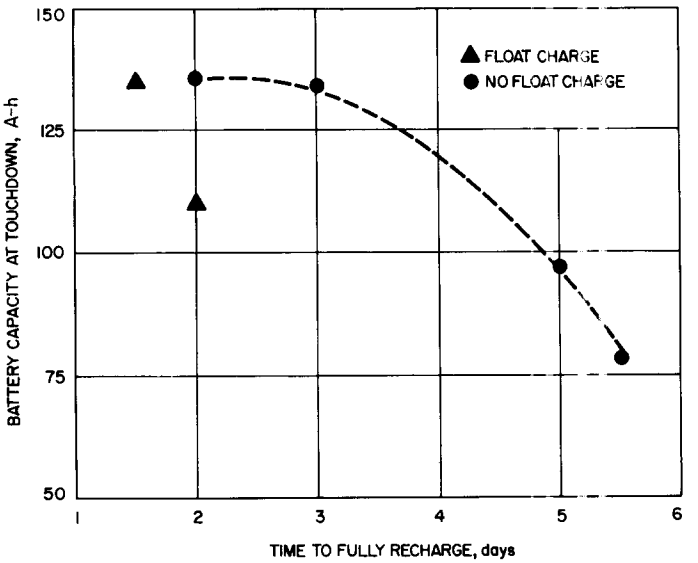

Fig. 15. Main battery capacity at touchdown vs time required to recharge

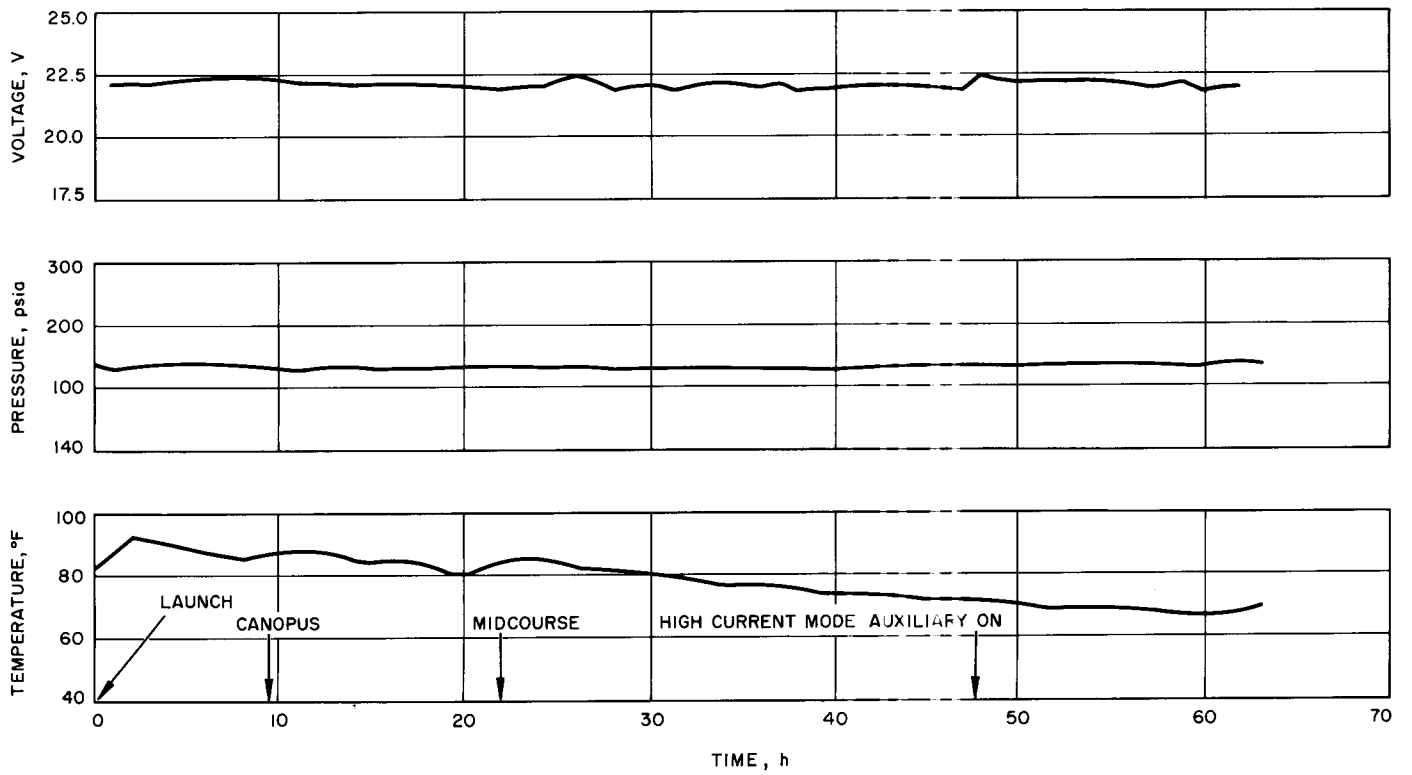

Fig. 16. *Surveyor III* flight data for main battery SN 108

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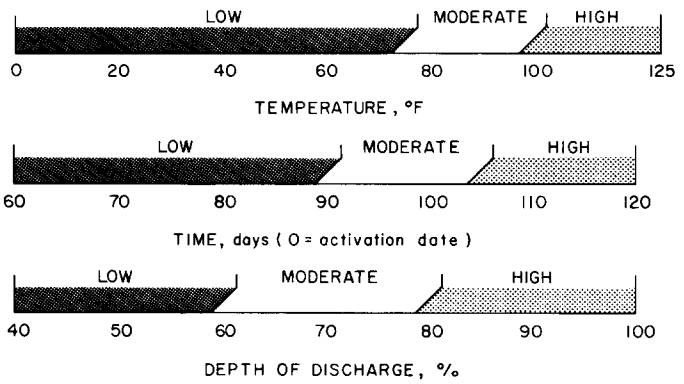
Battery failure analysis. During this report period, batteries SN 100 and 110 exhibited anomalous performances. Serial number 100 indicated an incipient shorting condition of decreasing cell voltages on three cells during open circuit stand. Serial number 110, after successfully completing an SC-5 STV mission, indicated an incipient shorting condition on one cell. Both batteries were considered failed when an individual cell did not accept a charge.

Evaluation of the events leading up to the failure of SN 100 showed that in the manufacture of cells 2, 3, and 4 (all of which exhibited incipient shorts, but only cell 3 did not accept a charge) the positive plates were from a different production lot than those in cells 1 and 5–14. On further examination, it was found that cells 2–4 had exceeded 1.96 V on float charging, while other cells were below 1.92 V. At voltages above 1.96 V, the evolution of oxygen is sufficient to promote premature degradation of those cells. Because of the clear relationship between positive plate lot differences and battery failure, corrective action was taken to eliminate variations between cells in positive lot material. The improved battery assembly method is described in detail in section 4A(1) of the Electric Storage Battery Company (ESB) subcontract, Positive Lot Control.



OPERATIONAL CONDITIONS	DEGREE OF RISK
L L L	VERY LOW
L L M	LOW
L M M	LOW MODERATE
M M M	MODERATE
H L L	MODERATE
H M L	HIGH MODERATE
H M M	HIGH
H H ⊛	VERY HIGH
⊛ ANY CONDITION	

Fig. 17. Main battery operational risk table

The failure of SN 110, although indicated by a failure of one cell to accept charge, was most likely caused by a combination of excessive use (approximately 133 A-h output) and high temperature exposure (122°F maximum). As a result, an operating risk chart (Fig. 17) was prepared to indicate the major factors contributing to battery degradation and the approximate degree of severity for each factor. Failure of future batteries may be reduced by attempting to operate batteries under conditions that are considered "low risk."

Revised pressure alarm limits. As a guide to safe battery pressure operating limits, a boundary curve of pressure and temperature was prepared for main battery operation in STV testing. The original boundary limit did not consider the effects of temperature in lowering the burst strength of the thermoplastic cell case material, in particular the cement joint strength. Tests conducted at ESB indicated a 50% reduction in bond strength at 125°F. Figure 18 was prepared as a guide for battery operation during STV testing. The previous operating guide did not designate a point at which action is mandatory; by adding such an action point it may be possible to avoid an abort condition. The operating level in any particular chart region is as follows:

Condition	Operating level
Safe	Up to specification maximums
Alarm	Low power mode only
Action	Discontinue battery use

b. ESB subcontract

Positive lot control. Engineering discussions with ESB resulted in a battery manufacturing plan to assure a more uniform product. Six batteries now constitute the minimum battery lot. Electric Storage Battery Company will fabricate each battery in the lot from a single production lot of positive plates. To do this, a new lot sampling method based upon revised plate weight categories is in effect. The average of a lot of 90 to 100 positive plate

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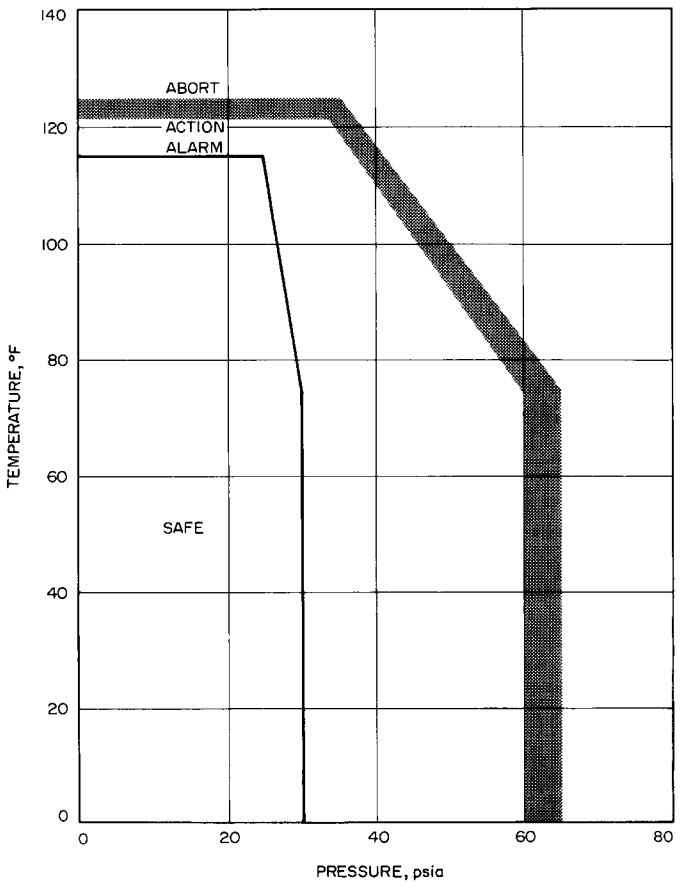

Fig. 18. Main battery operating limits

weights determines the sample mean of the sintered positive plate lot. The "nominal" plate category is equal to the mean weight plus or minus one-third within the acceptable weight range of 77.7 to 86.1 plate. The "light" plate category is equal to the lower one-sixth of the plates. A battery will be manufactured from a weight blend of six nominal plates in each cell or a blend of one light, four nominals, and one heavy plate in each cell. Previous batteries utilized any combination of acceptable positive plate lots necessary to meet cell weight requirements.

Selection of positive plates by the new weight categories has caused a wider variation in battery weight. Previously, the battery weight was regulated to a considerable degree by matching positive plate weights from more than one production lot. However, since selection is now limited to a single plate lot for a lot of batteries, the battery weight can vary directly with the mean plate weight of that lot.

Recently, several batteries have slightly exceeded the maximum specified weight of 46.5 lb. Electric Storage Battery Company has adjusted the positive plate process to produce a lower mean plate weight. Since this change will not affect current production batteries, the specified maximum battery weight has been raised to 46.8 lb.

Battery status. SC-4 flight batteries are being processed through FAT, and selection of the flight and two backup batteries will be made at the conclusion of these tests. The remaining three batteries fabricated for SC-4 flight will be designated as reliability test batteries.

c. Auxiliary battery

Surveyor III performance. Battery SN 57 was installed on *Surveyor III* for flight use. Figure 19 was prepared on a real-time basis from telemetry data. The plot was compared with data for the *Surveyor I* mission; based on this comparison, the battery performed satisfactorily.

3. Power Subsystem Test and Evaluation

Because of the unavailability of boost and battery charge regulators, no power subsystem group tests were performed during the report period. Three boost and two battery charge regulators remain to be tested during a recycle period for the spacecraft. The test report on group testing of BR-12/BCR-14 was prepared and distributed. These units were tested during the last report period. Also, test reports 3, 4, and 5 on similar numbered 71-h transit sequence tests were completed and distributed.

F. Telecommunications

1. *Surveyor III* Postlunar Landing Telemetry Analysis

Coincident with *Surveyor III's* second lunar touchdown, the signal processing of analog data became very noisy and unintelligible. Investigation indicated a failure of at least one commutator switch in both the engineering signal processor (ESP) and the auxiliary engineering signal processor (AESP). Analysis of telemetry data in all bit rates indicates the loading effect of the shorted commutator switches to be capacitive and, therefore, bit-rate sensitive. The group test console, which simulates all signal processing inputs and outputs, is being used to reconstruct the problem.

Based on empirical data supplied by the group tester, an ECAP computer program was developed to simulate the ESP/AESP commutator. Out of a total of 188 commutator switches, the suspected failed switches have been reduced to two of a possible five.

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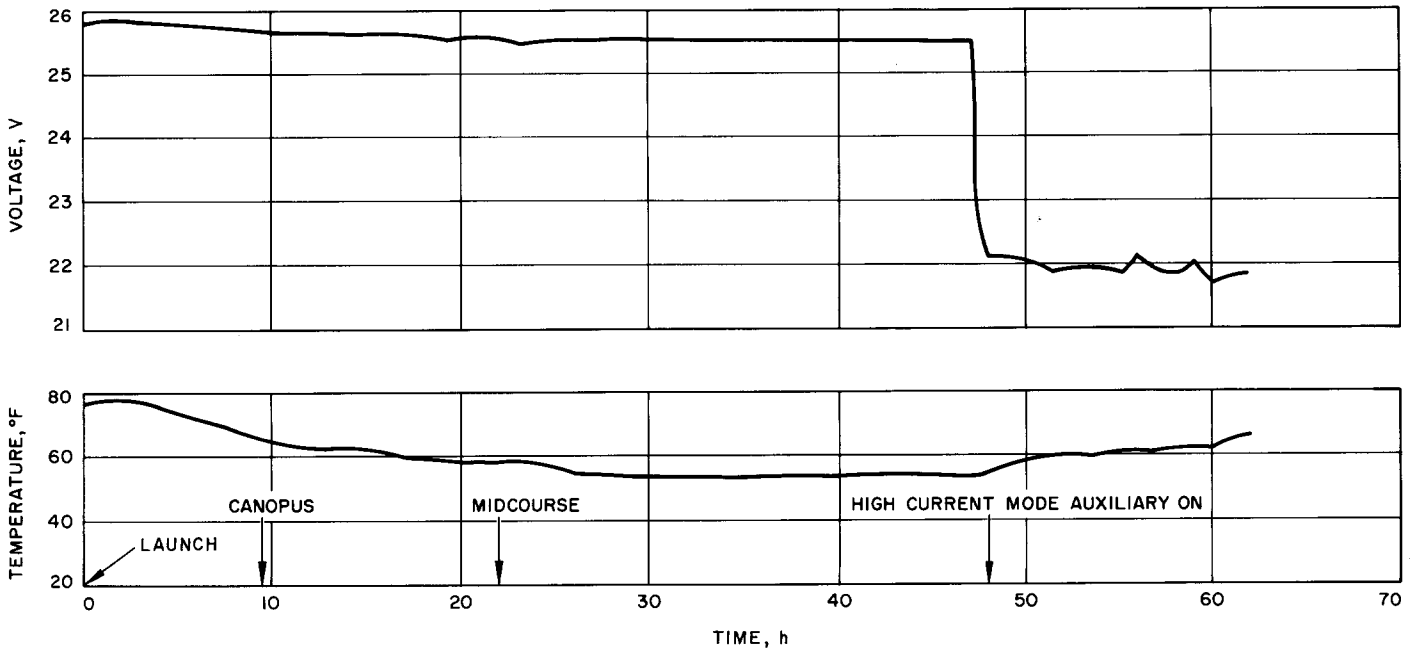

Fig. 19. *Surveyor III* auxiliary battery SN 57 flight data

2. Trouble Failure Report (TFR) Closures

Trouble failure report closures are continuing at a rapid rate. During the report period, the number of outstanding production TFRs was reduced from approximately 200 to less than 70. Mission-critical TFRs now total only 12, of which 6 are omnibus TFRs that must remain open for varying periods of time.

3. Altitude Marking Radar

The SP-1 altitude marking radar test evaluation was completed during this reporting period. The power fall-off problem encountered during thermal vacuum test was resolved by adding heaters to the test coupler.

The SC-6 altitude marking radar serial number SN 13 was returned to the unit area for boresight verification. During subsequent functional testing, an IF bandpass problem required reselection of the crystal and replacement of the IF amplifier.

Of vendor items, the RF assembly and modulator are the main areas of concern on current units (SN 12 and 14). Failures occurring during preliminary tests on the SC-7 unit (SN 12) have resulted in the tear-down of SP-2 (SN 14) for replacement parts. The modulator problem is expected to be solved in-house, but the RF assembly failure will require extensive investigation of both HAC and vendor measurement techniques. Two HAC engineers will visit the vendor's plant in order to correlate these measurement techniques.

4. Transmitter

Investigation of transmitter phase jitter continued during this reporting period. Instances of phase jitter at the system test level have increased concern for the subject in the unit area. Transmitter SN 15 has been returned for special testing because of this condition and is currently undergoing tests. A failure of transmitter SN 16 has resulted in an extensive investigation in the unit area for cause, overstressed parts, etc.

Type approval testing of the new configuration transmitter (3024400-1) has been completed. Deviations from specification requirements include an out-of-specification wideband frequency and an observed momentary drop-out in high power during vibration. Vibration rerun indicated this latter condition was apparent, not real; however, a substantial increase in phase jitter was noted. The excessive phase jitter has been traced to various manufacturing defects in the A 1 module. Transmitter SN 13 (the type approval test unit) will be revibrated during the next report period to verify the repair. The module (A 1) has already been revibrated.

5. Manufacturing

The last control item of signal processing equipment, an engineering signal processor (ESP) unit for SC-7, was

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delivered during April. One engineering signal processor serial number (SN) 5 unit was returned to the unit area for incorporation of an engineering order, with anticipated redelivery on June 9.

Transmitter deliveries continued at a fairly smooth pace during this reporting period. Transmitter 19 is expected to be delivered by June 10, leaving only one (SN 16) yet to be delivered. Expected delivery for SN 16 is June 23.

Altitude marking radar SN 14 (SP-1) was delivered, leaving SN 12 (SC-7) and SN 14 (SP-2) to be completed. Altitude marking radar SN 12 is currently in functional tests. Serial number 14 (SP-2) is short major subassemblies which had to be used as replacements for failures on SN 12.

Planar array antenna SN 21 is in top assembly stages progressing towards a June 30 delivery.

Retrofit of radar altimeter and doppler velocity sensor on-off switches has progressed on schedule. The first two relays have been delivered.

Receiver SN 28 was returned to the unit area for resweeping. After two modules were replaced (improvement changes), it was returned to systems test.

The major task under this work item for the next reporting period is completion of the spares requirement.

G. Payload

1. Soil Mechanics/Surface Sampler (SM/SS) Subsystem

a. SM/SS payload systems integration. The addition of an SM/SS to SC-7 requires spacecraft changes to add the SM/SS substructure and mounting hardware, and to provide a wiring harness to accommodate SM/SS power, command, and telemetry requirements. The SP-1 SM/SS subsystem, consisting of a mechanism, auxiliary, thermal compartment, and wiring harnesses, will be used. No unit spares will be provided.

The basic SC-7 spaceframe will be modified by adding the TV-4 substructure, an SM/SS mechanism mounting adapter, and an SM/SS auxiliary mounting adapter. The required TV-4 substructure and auxiliary mounting adapter will be removed from T-21 and used for SC-7. A new mechanism mounting adapter will be designed and fabricated to change the mounted azimuth orientation of the mechanism. This change is required to reorient the SM/SS operational area to encompass the alpha scattering deployment area, permitting repositioning of the alpha scattering sensor head.

The addition of the SM/SS adds approximately 16.7 lb of dry landed weight to the spacecraft. The new spacecraft CG location is well within acceptable limits. Actually, without the addition of the SM/SS, the Y-axis CG position was close to its maximum limit. Adding the SM/SS improves this condition by moving it closer to the Z-axis. Although several structural and thermal aspects require detailed study, no significant structural or thermal problems are expected.

Spacecraft wiring harness changes are required to provide the SM/SS with the following: (1) 22-V, 29-V, and power return leads; (2) four command lines; (3) an engineering mechanisms auxiliary (EMA) heater switch; (4) two digital, one analog, and three temperature telemetry channels, and (5) inhibit lines to the central signal processor master switch and to the TV auxiliary.

To avoid the cost of pouring another main retroengine, the SC-7 approach velocity requirement will be reduced to 2642 m/s, resulting in a decrease of the minimum post-touchdown Goldstone view period from 5.7 to 4.6 h in January 1968, and from 7.6 to 4.6 h in February 1968. However, it permits using the same retroengine as a spare for both SC-6 and -7.

To accommodate the additional payload weight, the SC-7 maximum midcourse capability will be reduced from approximately 42 to 39 m/s. The minimum required midcourse correction remains approximately the same, 10 m/s.

The addition of an SM/SS will require approximately 335 W-h (15.5 A-h) of main battery energy during the transit phase for the SM/SS auxiliary thermal compartment heater. This is not sufficient to affect battery terminal voltage during the descent phase. It does reduce the main battery energy margin at touchdown to 1700 $\pm$500 W-h (79 $\pm$23 A-h). This reduces predawn landing survival time, although the 20-h requirement can still be met.

The overall system test plan for spacecraft change verification will consist of approximately four shifts of penalty tests after the wiring harness changes are incorporated. There will be roughly two shifts of power and grounding

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retests and two shifts of spacecraft functional retest. The complete SM/SS subsystem will be added prior to spacecraft solar thermal vacuum test so that the thermal and structural, as well as functional integrity, will be verified during the normal system flight acceptance test program.

SM/SS subsystem operability on the spacecraft will be validated by a compatibility test prior to solar thermal vacuum, a functional test during STV, pre- and postvibration functional tests, and an SM/SS calibration test at the Air Force Eastern Test Range (AFETR). The SM/SS test plan is essentially the same as that conducted for SC-3 and -4.

In the 2 wk since contractual direction was given HAC to proceed with installation of the SM/SS on SC-7, the basic bus has been retrofitted to incorporate the addition of TV-4 substructure and the necessary spacecraft harness modifications.

Also during the April/May reporting period all control items were delivered and all testing scheduled under the original task was completed. At the spacecraft level, serial number 2 successfully completed special test 224B at AFETR.

b. SM/SS, SN 2, SC-4/AFETR calibration test. In accordance with change order 148, a special force calibration test was conducted on the SM/SS at AFETR. The test was scheduled for May 5 but was rescheduled to May 12 at JPL's request. (The rescheduling was requested because JPL personnel were engaged in SC-3 operational data reduction.) Special test plan 224B was written and approved, and a tape was prepared by May 1. All test personnel were at AFETR May 11, but the test was rescheduled by the SC-4 spacecraft manager to May 13. The test was initiated at 0630 h May 13 and was successfully completed by 1900 h. The force versus current data obtained compared very favorably with the predicted results.

c. SM/SS sustaining engineering. The sustaining effort has centered on the analysis of the motor test data that have been collected. In addition to this analysis, sustaining effort included support of the calibration test at AFETR and review and release of section 10, SM/SS subsystem requirements of the *Surveyor* spacecraft equipment specification 224832.

2. Television

Much of this report period has been devoted to type approval testing (TAT) II and evaluation which were completed in early June. Due to minor problems, retests were conducted to evaluate conclusions drawn from the following anomalies:

(1) Camera vidicon cathode degradation was determined to be a natural phenomena of the tube. All other tubes in flight hardware were carefully evaluated for a similar phenomena and found satisfactory.

(2) A bracket holding the iris potentiometer became loose. This was attributed to improper torquing and not to a design problem. Investigation disclosed all other units were satisfactory. A retest with proper torquing was conducted on the TAT-II camera and found to be satisfactory.

(3) The TAT review committee has declared the TAT-II results satisfactory, thus completing their efforts.

Overvibration SN 13 camera (SP-2) has been completely evaluated by a failure review board. After completion of rework, evaluation test, and a final flight acceptance test, the camera was accepted for flight use.

Engineering is currently evaluating possible modifications to SC-5 and subsequent spacecraft cameras to correct a transmitter overdeviation problem caused by the vidicon output voltage under hard saturated conditions. This condition could cause picture degradation and/or RF interference with other JPL programs.

H. Propulsion

1. Main Retroengine

a. Lot acceptance test (A22-10). Lot acceptance test engine A22-10 was successfully test fired April 14. The engine was temperature conditioned to the expected flight thermal gradient prior to test. Performance was nominal, and the t_{3500} prediction of 42.85 s was within 1% of the actual 43.2 s. The postfiring hardware condition was normal.

b. Surveyor III. The SC-3 retroengine A22-8 operated normally during the *Surveyor III* soft landing April 19. Preliminary data indicate the actual t_{3500} to be 41.0 s versus the predicted value of 41.09 s.

Retroengine performance was approximately 0.2% lower than the predicted specific impulse 289.5 lbf-s/lbm.

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Spacecraft pitch, yaw and roll requirements during retroengine burning were well within the vehicle control capabilities.

c. SC-4. The SC-4 flight retroengine A22-9 was prepared for flight while serving as the SC-3 spare engine. The safe and arm (S&A) device has been rechecked and found acceptable.

The SC-4 spare retroengine A22-11 was received at the Air Force Eastern Test Range (AFETR) May 15. It was subjected to an S&A check and X-ray examination. X-ray revealed an anomaly in the aft insulation boot. Thiokol Chemical Corporation (TCC), upon reviewing the X-rays, confirmed that a bulge in the aft boot existed which brought the boot in contact with the case wall insulation. Analysis and test are continuing to determine the engine's acceptability. As a result of the uncertainty of engine A22-11, and in order to protect the SC-4 schedule, TCC was requested to complete work on engine A22-12 and ship it to AFETR as a replacement spare retroengine.

d. SC-5. The SC-5 flight retroengine A21-28 is in the AFETR magazine awaiting final preparation for flight use.

The SC-5 spare retroengine A21-29 has been accepted by HAC at TCC and is scheduled for shipment to AFETR late in June.

Both of these A21 retroengines are loaded with 1250-lb propellant.

2. Vernier Propulsion System

a. System analysis. Propulsion support was maintained for the entire *Surveyor III* mission. Since the landing, propulsion data have been monitored to assess propulsion system condition in case further actuation is planned regarding static firing or hopping.

Effort is under way to update all propulsion documentation for the SC-4 mission. New formats have been generated to take advantage of the additional real-time display capability provided by the reprogramming of the 7044 computer system at the JPL Space Flight Operations Facility (SFOF).

b. Spacecraft servicing

SC-5. A leak in the vicinity of the leg-3 thrust chamber assembly (TCA) was noted during solar thermal vacuum (STV) testing while the vacuum chamber has chilled down (walls at −320°F) and under a hard vacuum. The test was aborted and the vehicle returned to ambient conditions, at which time the TCA was inspected. No leakage was visible. The pad pressure was vented, and the fuel and oxidizer check and relief valve assemblies were changed due to being exposed to temperatures in excess of the specification requirements during STV testing. Troubleshooting sequences to determine the leakage consisted of the following:

(1) Repeat pressure decay test at 365 psig.

(2) Leak detector test (portable Halogen leak detector).

(3) Reload (topoff) tanks to determine amount of fluid missing.

(4) Repressurize and perform residual gas analyzer test.

(5) Off-load alcohol tank legs 2 and 3 and vacuum dry for 12 h, then pressurize at 35-psig helium and repeat Consolidated Electrodynamics Corporation (CEC) leak detector test.

(6) Chill down TCA to temperature at which the leak occurred and repeat the CEC test with 35-psig helium.

(7) Reload the system and repeat the pressure decay test.

(8) Return to STV and repeat the eclipse portion of STV with mirrors placed to view the rear of the TCA.

Items (1) through (7) did not indicate any leakage from the vernier propulsion system. Item (8) resulted in visual observation of the liquid leaking from the bellows cover plate of the TCA throttle valve.

The leg-3 TCA was changed and, during the troubleshooting process, a leak was detected in the leg-2 fuel tank, requiring this tank to be changed. A pressure decay test was then performed with no problems noted. Following STV, the tanks were subjected to a special bladder leak test. This consisted of filling the tanks full and, while maintaining vacuum on CV-3 and -4, pressurizing the liquid side of individual tanks at 20 psig for 1 h/tank. A high-pressure leak test was then performed by pressurizing the system at 750 psig and observing for leakage. No leakage was noted as a result of either test. All pressures were then set at 250 psig in preparation for the vibration test.

SC-6. The dummy TCAs were removed to allow a strain gage calibration to be performed and the flight engines

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to be installed. The propulsion system was serviced in preparation for STV.

SC-7. The leg-1 oxidizer tank was installed on the vehicle. The leg-2 fuel tank was removed and is presently installed on SC-5. The dummy TCAs were removed to allow a strain gage calibration to be performed and were then reinstalled.

c. Propellant tanks. The bladder from the failed oxidizer tank 287119, SN 3, which leaked at AFETR while installed on SC-3, was analyzed by Dilectrix. The cause of the blisters was an improperly applied layer between the Teflon (TFE) and FEP (inner layer).

The fuel tank removed from SC-5 was vacuum dried and then subjected to a bladder leak test. The leak rate was too high to be measured on a bubbleometer. The tank was subsequently disassembled, and examination of the bladder revealed two cracks in the outer layer of FEP with pinholes in the TFE in the area of the cracks. Both cracks were in the lower part of the bladder. Cause of failure is being investigated.

d. Check and relief valves. On SC-5, the check and relief valves were exposed to 170°F during STV testing.

Because of this out-of-specification test condition, 125°F maximum versus 170°F exposure and the extended time at the high temperature (approximately 70 h), a special test program was undertaken to determine change in valve performance, if any.

The tests in the following sequence were performed at 70, 125, 150, and 170°F.

(1) Check valve leakage at 4 and 300 psig.

(2) Relief valve cracking pressure and reseat pressure.

(3) Relief valve leakage at 300 and 790 psig.

(4) Overall leakage of valve assembly at 300 and 790 psig.

(5) Check valve opening and flow capacity.

Before the valve tests at 150 and 170°F, the valves were exposed to isopropyl alcohol for 3 h.

During the 150 and 170°F tests, the valves were pressurized to 300 psig to simulate pad pressure on the spacecraft. This pressure was applied to port B, and the valves were soaked at elevated temperature for a minimum of 14 h before their functional tests.

Results of the test were within specification up through 150°F. At 170°F, the reseat pressures were below specified (125°F) values (specification: 800 psig minimum; actual: 796 psig).

3. Manufacturing

During this report period, six propellant tanks were completed and two other propellant tanks were reworked after bladder failures.

Because of the rejection of the holeless standpipe assemblies, a new weld schedule was developed and six new standpipe assemblies completed.

A check and relief valve overtemperature on SC-5 required rework of SN 16 check and relief valve 287290-50 and the flight acceptance testing of this assembly, which was completed.

During STV testing of SC-5, the check and relief valves were exposed to an out-of-specification high temperature condition, and a test program was completed to determine the change in valve performance. This test program required the assembly of two new valves, one 287290-30 and one 287290-50, including the flight acceptance functional testing, which was completed.

4. Thrust Chamber Assembly[1]

The program for *Surveyor* vernier engine [Reaction Motors Division (RMD)] Model No. TD-339) evaluation continued with eleven engine firing tests and miscellaneous flow and storage tests.

a. Engine solvent and fuel storage tests. The final phase (tests F-31 and -32) of the fuel storage program, involving storage of an engine with solvent followed by fuel, was completed on March 14, 1967. Two solenoid valves SNs 207 and 211, each rinsed with a film of solvent followed by fuel, accompanied the engine in the storage program. Significant results are as follows:

(1) Only a few drops of either isopropyl alcohol or MMH · H_2O were found in the helium actuation line following 2 wk of storage with solvent and 3½ wk of storage (20 days at 75°F and 4 days at 100°F) with fuel.

(2) Engine valve timing tests, performed before and after fuel storage, indicated (for the opening cycle) an increase of only 10 ms in the first actuation and no change in subsequent actuations. There was no change

[1]Prepared by JPL Technical Section 384.

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in timing for the shutdown cycle. Valve timing was within specification.

(3) The presence of fuel in the two solenoid valves affected their operability following storage and subsequent drying. In the case of SN 207 valve, it would not actuate (poppet shuttle) at 30 V. Upon disassembly, a heavy black residue was found on the internal parts. Analysis showed it to be nickel oxidized in the presence of air and complexed by the monomethyl hydrazine. In the case of SN 211 valve, the opening time was increased by a factor of two following storage; closing time increased by a factor of 2 to 4 (3 timing tests). After exposure in a special test to an environment of 130°F, the poppet failed to shuttle at 20 V. Since this valve had been subjected to the same storage conditions as the SN 207 valve, the presence of the same residue was suspected. Following a water flush to dissolve the residue, it operated properly with a normal 16-V applied voltage.

In summary, the solenoid storage tests indicated that if there is any possibility in the spacecraft system of fuel vapors penetrating to the solenoid valve, the nickel plating may be complexed by the fuel. Normally no fuel can get into the cavity until the first firing, which occurs in space with no air for complexing. The test results emphasize the need for a good leak check of the pressure seat of the solenoid valve; this is normal AFETR procedure.

The engine storage test performed under simulated spacecraft ground storage conditions indicated that only a few drops of fuel entered the helium actuation line. The opening and closing cycles following storage were essentially unaffected by contact with the solvent and fuel.

b. Engine solvent and oxidizer storage tests. The storage program described in subsection 3*d.* of SPS 37-44, Vol. I, p. 26, was initiated February 15, 1967 and completed March 14. One gold-plated sea-level engine (SN 525) was tested with Kel-F lubricated seals; the other engine (SN 523) was tested with selected seals lubricated with DuPont PR 240 AC. Inspection of the two engines following the final 5.6-day[2] storage period, under pad pressure with oxidizer, indicated no evidence of oxidizer seepage for the Kel-F lubricated, SN 525, engine. Two seepage areas developed in the SN 523 engine; one at the Kel-F lubricated bellows end cap and the other at the DuPont lubricated throttle valve filter snap ring. The seepage products were a white, fine, crystalline material and were similar in appearance to the products observed on SC-1 and -2 engines at AFETR. An infrared spectra analysis of samples of the material matched that of a metal nitrate, with no evidence of fluorocarbons in any of the spectra.

A review of the inspection records at AFETR for SC-1, -2, and -3 engines indicates that oxidizer seepage is related to the leak tightness of the individual joint as measured by a Consolidated Electrodynamics Corporation helium leak detector probe. In general, when the leakage rate of the tracer gas (99% N_2, 1% He) at 300 psi exceeded about 100 cm^3/h (within the specification limit of 200 cm^3/h at certain seal areas), seepage products became apparent. This was particularly so in the case of SC-1 and -2 engines. For SC-3 the highest leakage rate prior to oxidizer loading was 13.7 cm^3/h. A small amount of seepage was noted during storage at AFETR in one area only, the oxidizer bellows end-cap omniseal.

In the subject storage program, seal leakage rates for the two engines could not be determined with a helium leak detector probe for lack of the type of equipment used at AFETR. Using standard test station procedures (helium gas and soap solution at joint), however, slight bubbling was noted in the area of the oxidizer filter (a seepage area) of SN 523 engine during leak test. No leakage was apparent in the other seal areas. Again, the test data would appear to indicate that the development of oxidizer seepage products is related to the leak tightness of the joint.

It may be concluded that the type of lubricant (Kel-F or DuPont PR 240 AC) used is not a controlling factor in the development of oxidizer seepage. Whether the rate at which seepage products develop is a function of the amount of lubricant applied to the seals was not established in the subject program.

c. Injector salting. The test program described in subsection 3*e* of SPS 37-44, Vol. I, p. 26, was continued with four additional tests (G-38−B to −E). Table 12 identifies the environment to which the engine was exposed in each test and indicates whether or not salting took place.

Tests G-38—B and —C were performed with no air flow by a sea-level engine at a dry bulb temperature of 90°F. With 103 gr of moisture (corresponding to a relative humidity of 49%) partial salting took place after 24 h of storage. With reduced moisture (58 gr and a relative humidity of 27%) no salting took place.

[2] In simulation of the time at AFETR for oxidizer seepage to develop (March 8 to 14).

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Table 12. *Surveyor* engine salting tests at AFETR (TCA SNs 522, 532 and 507)

Test G-38	Air flow velocity, ft/min	Environment						Injector[b] salting
		Grains H_2O/lb dry air	Relative humidity, %	Temperature, °F		Duration, h	Average MON-10 leakage rate, ml/h	
				Dry	Wet			
-A, -G through -Z[a]								
-B	0	103	49	90	74	24	0.10	Partial
-C	0	58	27	90	66	24	0.08	No
-D[c]	15	210	100	90	90	24	0.12	Yes
-E[d]	10	56	42	75	60	120	1.7	Partial

[a] See Table 5 of SPS 37-44, Vol. 1, p. 28, for test results.
[b] Yes—complete plugging; No—no plugging; Partial—some plugging.
[c] Flight nozzle taped to sea-level engine nozzle.
[d] Test performed with TAT-engine SN 507.

To this point in the program, all tests had been performed with a sea-level engine. A question was raised as to whether the presence of a flight nozzle would influence salting for those cases involving air flow (aspiration effect) by the engine. Test G-38—D was performed with a sea-level engine (SN 522), to which a flight nozzle was attached by Teflon tape. Test conditions were identical to those for test G-38—Z (210 gr of moisture; 100% relative humidity; 90°F dry bulb; 20 ft/min air flow) performed with a sea-level nozzle, wherein no salting took place. As noted in Table 12, 100% blockage of the injector took place in test G-38—D. Apparently the presence of a flight nozzle minimizes the benefits realized from air flow by a sea-level nozzle. During the course of the experiment with a flight nozzle, the presence of MON-10 vapors within the chamber could be observed with the aid of a mirror held at the exit of the nozzle.

The final test (G-38—E) in this series was conducted with type approval test engine SN 507, which had no prior history of salting. Test conditions were chosen to represent that period within the shroud when the nozzle plugs have been removed, and the engines may be exposed to limited humidity for a length of time up to 5 days (simulating the possibility of a hold period of this duration). As indicated in the table, partial salting and blockage (oxidizer flow approximately 50% normal) of the injector took place after a 120-h storage period. The results of this experiment indicate that time is an important factor in the formation of salts in the injector. In several tests previously reported, salting did not take place after a 24-h storage period under conditions involving a larger moisture content and higher temperature than was the case in test G-38—E. Laboratory tests, however, have indicated that nickel will react with MON-10 to form salts at a relative humidity as low as 32% at a dry bulb temperature of 72°F. Salting was observed to start after only a few hours of exposure of the material sample to MON-10 fumes. Thus, the engine data reported herein and in SPS 37-44 are not inconsistent with the laboratory results previously discussed.

The subject program, while limited in scope, has served to define the effect of many of the variables affecting salt formation in a nickel injector. Major variables investigated were MON-10 leakage rate, moisture content, temperature, duration of exposure and air flow by the engine. Conclusions as to the effect of each of these variables are briefly as follows:

(1) MON-10 leakage rate through the shutoff valve seal is undoubtedly the most important single variable affecting salt formation. Tests of two normal leak-tight engines for 20 and 33 days under a simulated AFETR environment have indicated minimum NO_2 leakage (0.7 mg/day) to the injector, no change in leakage rate with time, and no evidence of salting. Coupled with this is the fact that in over 200 design verification and TATs at Edwards Test Station (ETS) over a wide range of conditions and storage times, there has been no evidence of leakage across the oxidizer poppet seat of the shutoff valve (except for two or three cases involving abnormal test conditions). In several tests of an engine over a 24-h period under leak rates several orders of magnitude larger than normal, and within the limiting environment (when encapsulated) outlined in JPL project document PD-1, injector salting did not take place.

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(2) Moisture content, temperature, and relative humidity level are primary factors influencing salting. The data from tests G-38—Q, —S, —W and —Y (SPS 37-44, Vol. I, p. 28), in which the average MON-10 leakage rate to the injector was 0.40, 0.27, 0.08 and 0.09 ml/h, respectively, indicate that for over a 24-h period salting does not take place when the moisture in the air is less than 103 gr/lb of dry air and the dry bulb temperature is less than 80°F. Data from test G-38—C (average MON-10 leakage rate of 0.08 ml/h) indicate that over the same period of time salting does not take place at a moisture level of 58 gr and a dry bulb temperature of 90°F. The data from tests G-38—O, —P, —T, —U, —X and —Y (SPS 37-44, Vol. I, p. 28) indicate that at higher moisture contents (ranging from 103 to 210 gr) and with a dry bulb temperature of 90°F, either partial or complete blockage of the injector orifices took place over a 24-h storage period. The initial MON-10 leakage rate to the injector in tests G-38—O, —P, —T, and —X was 0.09, 2.6, 0.12 and 0.17 ml/h, respectively. The average MON-10 leakage rate in tests G-38—U over a 18-h period was 0.19 ml/h. The average MON-10 leakage rate in test G-38—Y over a 24-h period was 0.09 ml/h. Initial MON-10 leakage rates are given for those tests in which salting took place and blocked the flow. Average MON-10 leakage rates are given for those tests in which either "no salting" or "partial salting" took place.

(3) The effect of exposure duration upon salting has been indicated previously in the discussion of test G-38—E test results. Given a sizeable MON-10 leak to the injector and sufficient time, salting will take place at a moderate temperature (75°F) and moisture level (56 gr). When temperature and moisture level are increased, chemical attack will proceed faster and either partial or complete blockage can take place within 24 h.

(4) Initial tests with a sea-level engine indicated that air circulation past the engine had a pronounced effect in the prevention of salt formation. The data of tests G-38—O and —Z (SPS 37-44, Vol. I, p. 28) demonstrated that air flow by the nozzle (20 ft/min velocity) will prevent salting at a relative humidity of 100% (corresponding to 210 gr of moisture) and a dry bulb temperature of 90°F. At a later date, test G-38—D was performed with a flight nozzle attached to the engine under the same set of conditions for test G-38—Z. As indicated previously, complete blockage of the injector took place. It is apparent that air flow by a flight engine, in itself, cannot be counted upon to prevent nickel salt formation.

d. SC-2 oxidizer restriction tests. Included in the SC-2 failure analysis tests reported in SPS 37-43, Vol. 1, p. 7, was a group of tests (DD-178 to -183) in simulation of the possible failure mode during the second burn of the leg-3 engine. The tests were performed with a shutoff valve which had been modified to restrict the oxidizer poppet movement by predetermined amounts, since analyses had indicated that there was a restriction in the oxidizer flow circuit of the leg-3 engine. The thrust data from test DD-182 provided the closest simulation to the spacecraft-telemetered strain gage thrust information, which indicated about 2 lb of thrust for about 5 s. In test DD-182 the engine developed a peak thrust of about 5 lb for 1 s; thrust then fell to about the ½-lb level for the remaining 5-s period. The engine operated at a mixture ratio under 0.036.

Tests DD-178 to -183 had demonstrated the characteristics of an engine when flow is restricted by limiting the movement of the oxidizer poppet. Tests DD-223 to -226 were performed to explore the effect of a restriction in other places in the oxidizer flow circuit. The restriction was provided by a needle valve adjusted to provide an oxidizer flow rate of about 0.007 lb/s at a throttle valve setting of $+80\Delta$ mA. Test duration was 2 s in simulation of SC-2 burn No. 2. The engine was mounted from flight brackets. The engine, oxidizer line, and propellant supply were temperature conditioned prior to each firing to spacecraft values. The first two tests (DD-223 and -224) were performed with the restriction located at the throttle valve inlet. Oxidizer flow rates were higher than planned; the corresponding thrust was about 15 lb.

In test DD-225, the restrictor valve was placed in the oxidizer line leading from the throttle valve to the shutoff valve. Finer adjustment of the valve resulted in the development of the desired flow rates during the firing. Thrust spiked to about 12 lb for a 1-s period and then dropped rapidly to about 2.5 lb, gradually decaying to zero by 5 s. Test DD-226 was performed with the restrictor valve mounted again at the throttle valve inlet and with no change in adjustment from the prior test. The maximum thrust developed was about 8 lb. The rate of thrust build-up and decay, however, was slower than in test DD-225, eliminating the thrust spike that developed when the restrictor was located immediately upstream of the shutoff valve.

In summary, tests to date with oxidizer flow restricted at three different locations have all exhibited an initially high thrust followed by a reduction to a low level. Flight data (based on several attempted 2-s burns), on the other hand, has consistently indicated no thrust spike and a uniformly low level of thrust (about 2 lb) lasting for

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about 5 s. The development, of a thrust spike in the restriction test can be understood when one considers the delayed entry (resulting from the reduced oxidizer flow rate) of oxidizer into the injector manifold combined with the fact that a relatively large quantity of fuel is flowing into the combustion chamber in a normal time sequence. Ignition may be characterized as a "hard start" followed by a pseudo-equilibrium period at low thrust. These data would lead one to the conclusion that the most likely point of restriction which could produce a thrust-time curve in simulation of SC-2 leg No. 3 engine must be located at the injector orifices. With a restriction at this point, both injector manifolds would fill at a normal rate, but thrust would be continuously low because of the incomplete combustion associated with the extreme fuel rich mixture ratio.

Attempts will be made during the next report period to subject an engine to a salting environment and induce plugging of the injector to the point that the flow rate which can pass through the orifices will be about 0.007 lb/s. A firing test will then be performed as a continuation of the subject program.

e. Solenoid valve tests. A detailed investigation of the flight worthiness of the solenoid valve assembly was initiated by JPL in January. The valve is produced by the Valcor Corporation, Kenilworth, N.J. and is delivered to RMD of Thiokol for assembly to the engine. The valve has passed all qualification tests at the component, engine and propulsion system level. No difficulties have been experienced in engine design verification tests at ETS when the valve has been exposed to *normal* operating conditions. Following exposure to certain abnormal conditions, valves have either failed to operate within specification limits or have not operated at all. All solenoid valves functioned properly on SC-1 and -3. There is no reason to believe that the SC-2 leg No. 3 engine failure is related to solenoid valve operability. There was considerable concern prior to SC-3 launch that excessive valve temperature at the end of the retromotor descent firing might result in either an unduly long delay in engine shutdown or no shutdown at all. The results of the investigation into the characteristics of the valve leading up to the final decision that there would be a good probability of successful solenoid valve operation throughout the SC-3 mission are as follows:

The initial investigation was made with four nonflight valves (SNs 207, 138, 214, and 220), which had been subjected to considerable service at RMD and ETS. Inspection and actuation tests indicated: (1) minimum diametral clearances (0.001 in.) can exist between the plunger and the bobbin; (2) rust on interior parts; and (3) failure to pull in at 16 V. Based on these preliminary findings, the investigation was continued with three flight-type solenoid valves (SNs 251, 252 and 256) which had been acceptance tested at Valcor, sent to HAC for use in the STEA equipment, and subsequently returned to JPL. A fourth valve (SN 228), which had been acceptance tested on flight engine SN 551 at RMD with subsequent storage under normal conditions, was removed from the engine and sent to JPL for evaluation.

Chronological history of SN 228 solenoid valve:

(1) Acceptance tested at RMD with engine.

(2) Delivered to JPL with engine for joint conductance tests.

(3) Engine shipped to ETR for flow tests.

(4) Engine delivered to Hughes Aircraft Company (HAC), El Segundo, California.

(5) Engine installed on SC-3 during limited solar thermal vacuum testing.

(6) Engine removed from SC-3 and shipped via JPL to RMD for contamination investigation.

(7) Removed solenoid valve from engine and sent to JPL.

The evaluation program was to be one of both inspection and test. By use of a precision microscope, it is possible to measure the interior dimensions (ID and plunger clearance) of each valve body in the plane of the four outlet holes without valve disassembly. Following such inspection, the test plan called for a determination of the operating characteristics of each valve. Test variables were to be temperature (ambient, 100, 165 and 300°F) (one valve only), applied voltage (15.5 to 16.5 V) and helium actuation pressure (0, 720, and 800 psia). Valve opening and closing times were to be determined by photographing current, voltage and pressure traces on an oscilloscope.

Coil resistance as a function of temperature was to be determined. Following test, all valves were to be disassembled and inspected for cleanliness and design clearances.

Table 13 presents the opening time of several valves (SNs 251, 252, 256, and 228) at various temperatures. All valves passed a modified acceptance test in that, at

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Table 13. Solenoid valve tests at JPL (as received condition)

Date, 1967	SN	Temp, °F	Relative humidity, %	P-He, psia	Applied volts, dc	Opening[a] time, ms
Feb 24	251[e]	68	28	0	16.0	40
		60	↓	700	↓	27
		104		700		28
		60		800		25
		104		800		26
Feb 27	252[e]	64	28	0	16.0	77
		61	↓	700	↓	27
		68		700		30
		104		700		30
		168		700		33
		65		800		28
		103		800		28
		169		800		29
		300		800		—[b]
		67.3		700		39[c]
Mar 1	256[e]	68	26	0	16.5	—[d]
		64	↓	700	16.0	33
		104		700	16.0	48
		169		700	16.0	50
		242		700	16.5	51
		251		700	17.0	63
		62		800	16.0	31
		104		800	16.0	35
		169		800	16.0	42
		245		800	16.0	57
		251		800	17.0	53
Mar 2	228[f]	70	—	720	16.0	26
		70	—	720	26.0	13.5

[a] Time interval from voltage signal to plunger movement.
[b] Valve would not operate at 300°F.
[c] Leak at vent (following 300°F test) with valve in actuated position.
[d] Minimum pull-in voltage of 16.5 dc at P-HE = 0 psi.
[e] Valcor AT valves.
[f] RMD flight AT valve.

ambient temperature and with 700 psi helium actuation pressure, they all shuttled properly with an applied voltage of 16 V. Opening times (voltage signal to plunger movement) for two of the VALCOR-AT valves (SNs 251 and 252) varied from 25 to 39 ms in tests over a 65 to 165°F range and a 700 to 800 psi range. The opening time of the third VALCOR-AT valve (SN 256) varied from 33 to 50 ms in tests over the same range of conditions. Flight valve (SN 228) was only tested at ambient temperature in this series of tests. Its opening time at ambient temperature varied from 26 to 13.5 ms over a voltage range of 16 to 26 V dc, respectively. Data on closing times for the four valves are not presented in Table 13 because the data were obtained with the valve deenergized from 26 V. In later tests, shutdown times were obtained from the proper 23 V (RMD specification 20429 value) level.

It may be noted that SN 256 valve would not shuttle at 16.5 V and zero actuation pressure; with 700 psi helium pressure, the valve met specification requirements, which call for actuation at 16 V.

Two valves (SNs 252 and 256) were tested at temperatures well above the normal range. Valve SN 252, when heated to 300°F, would not actuate. In a subsequent ambient temperature test, the valve leaked gas from its vent when in the actuated position. Valve SN 256 operated successfully at a temperature of 251°F, although higher than normal (17.0 V dc) applied voltage was required. Opening time (63 ms) was about twice the maximum specified acceptable value (30 ms at ambient temperature).

Following these tests, valves SNs 228 and 251 were inspected March 20, 1967 for contamination by flowing a precision cleaning agent, freon-TF, through each valve and collecting particles on filter elements. Several hundred metallic particles and nonmetallic crystals of various sizes (up to 500 μm) were flushed from each valve. Following this contamination check, the valves (together with SNs 252 and 256) were candidates for further investigation.

Following the contamination check, the four valves were disassembled for dimensional inspection and the condition of the interior surfaces. All parts conformed to design dimensions. Rust was found on the poppet and bobbin of SNs 251 and 228 valves. Serial number 228 valve was reassembled in its rusted condition to determine whether rust would affect its operating characteristics at higher temperatures. The rusted parts of SN 251 valve were cleaned prior to reassembly. It was found that rust can be effectively removed with a rinse made up of 5% citric acid, which has been neutralized with an excess of ammonia. This solution does not attack metals but does complex iron as well as nickel, chromium, and aluminum compounds (i.e., oxides). Procedures for the removal of possible rust in an assembled valve have not been developed.

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Following reassembly, SNs 228 and 251 were tested for opening and closing response at elevated temperatures. Test conditions were in accordance with those outlined in RMD specification 20429 (i.e., 16 V dc to energize the solenoid and 23 V dc to deenergize the solenoid). Maximum acceptable opening and closing response times called for in this specification are 30 and 17 ms, respectively.

The test equipment included a volume which simulated that of the actuation cavity (in the actuated position) in the shutoff valve. Helium flow in and out of this volume was sensed by a pressure gage and recorded on an oscilloscope together with current and voltage traces. Table 14 presents timing and pressure response data for the opening and closing cycle of these two valves, as well as stroke measurements at ambient and elevated temperature.

The data for flight valve SN 228 illustrate the characteristics of a valve whose stroke (0.005 in. at ambient temperature) is considerably less than the design value of 0.010 to 0.012 in. at ambient temperature. While opening and closing response times are within specification limits, pressure decay time is considerably longer than that (45 ms) of a normal stroke valve.

Also to be noted is the fact that SN 228 valve would not actuate after it had been heated to 165°F. (As will be indicated later, such a temperature can be reached at the end of a simulated retroengine descent firing test.)

Serial number 251 valve was tested over an ambient to 165°F temperature range. The stroke of this valve is slightly greater than the design value. Opening and closing response times are within specification limits. Pressure rise and decay times would permit a rapid engine shutdown.

Comparing the data from these two valves, it may be noted that the opening time for short-stroke valve SN 228 is considerably less than that of the normal-stroke valve SN 251. Pressure rise and decay times, however, are considerably longer for the short-stroke valve. These overly long pressure rise and decay times are associated with the short stroke (0.0025 in.) of SN 228 valve at elevated temperature and the resulting restriction in

Table 14. Solenoid valve tests at JPL (following reassembly)

Date	SN	Temp, °F	Helium pressure, psia	Opening cycle			Closing cycle			Stroke, in.
				Volts, dc	Open[a] time, ms	Pressure[c] rise, ms	Volts, dc	Close[b] time, ms	Pressure[d] decay, ms	
Apr 1, 1967	228	78	720	16	17	41	23	12	85	0.005
		78	↓	↓	11	—	↓	—	—	—
		125			18	57		11	100	—
		145			18	84		10	275	0.0025
		165		No actuation						
Apr 6, 1967	251	75	720	16	35	45	23	10	35	0.0145
		125	↓	↓	35	48	↓	8	38	—
		145			36	50		8	42	0.010
		165			34	49		7	45	—
(As received condition)										
Apr 7, 1965	203[f]	63	720	16	26	37	23	14	45	0.010[e]
Apr 7, 1965	235[g]	63	720	16	24	34	23	10	45	0.009[e]

[a] Time interval from voltage signal to plunger movement.
[b] Time interval from voltage signal off to plunger movement.
[c] Time interval from voltage signal to full supply pressure in actuation cavity.
[d] Time interval from voltage signal off to full pressure decay in actuation cavity.
[e] Stroke same before and after five actuation tests.
[f] Solenoid from flight engine SN 537.
[g] Solenoid from flight engine SN 555.

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the gas flow circuit through the valve. These findings prompted further investigation to: (1) determine whether other flight type valves had less than normal stroke characteristics; (2) determine analytically and experimentally the principal factors associated with the change in stroke; and (3) determine the effect of valve internal heating on its stroke characteristics.

Two solenoid valves (SNs 203 and 235) were removed from flight engines and tested at ambient temperature. Each valve was subjected to four actuation cycles. The stroke before and after test was measured and found to be unchanged. Following the actuation tests, opening and closing response and pressure rise and decay measurements were made. As indicated in Table 14, the stroke and performance of these two valves was essentially within specification values.

Concurrently with the above testing, the plungers from two valves (SNs 252, 207) were tested to determine their growth when heated to 165°F. The tests were performed by holding the plunger in a V-block, and obtaining a reference ambient temperature dial indicator reading of the height of the nylon at one end of the plunger while the other nylon end rested on a special fixture on a surface plate. The plunger and V-block were then put in an oven, held at 165°F for 1 h, and then returned to the original position on the surface plate fixturing. It was possible to obtain a new dial indicator reading within 20 s of removal of the plunger from the oven. These tests showed a growth of 0.0065 and 0.0075 in. in the plunger of SN 207 and 252 valves, respectively. The major part of the growth was in the nylon insert material and there was some variation in the amount of such growth. From analysis, it would appear that the amount and variability of the growth is a function of the fit-up (press-fit at one end and threaded joint at the other end) of the nylon insert within the metal plunger.

To determine the amount of heat that flows to the plunger (and the difference in temperature between the plunger and the bobbin) when the valve is held in an actuated position for 5 min, fine wire thermocouples were welded to the metal walls of the plunger and bobbin of SN 256 solenoid (previously disassembled). The solenoid was then installed inside a bag and filled with helium to provide a helium atmosphere at atmospheric pressure within the valve. The heat transfer from the bobbin to the plunger through the gas would then be properly simulated, since thermal conductivity of helium is essentially unaffected by pressure. The tests were not run in vacuum since the major purpose of the experiment was only to determine the difference in temperature between the plunger and the bobbin. Applying normal voltage to the valve for 5 min, the temperature-time history of the plunger and bobbin was determined. The temperature rose from 75 to 120°F, the plunger being about 2°F hotter than the bobbin. The major conclusion to be drawn is that the temperatures of these parts are essentially the same within the accuracy of the measurements.

Pursuing temperature rise effects further, bench flow tests of an engine were performed to determine the temperature rise history of the solenoid valve, with and without solvent flow through the engine shutoff valves. It was known that continuous application of electrical power to the valve would cause a temperature rise of about 45°F in 5 min. The purpose of these tests was to determine the degree to which fluid flow through the throttle valve and shutoff valve would affect the thermal behavior of the solenoid valve. The tests were run in simulation of conditions (i.e., flow rate and duration) which might prevail on SC-3 retroengine descent. It was recognized that the simulation would not be exact, since it was not practicable to heat the isopropyl alcohol to the fuel outlet temperature. Solenoid valve temperature could be derived from a measurement of the change in resistance in the solenoid armature during the 200-s flow test. Two tests were performed, one with solvents filling the valve cavities but no flow, the other with solvent flow through the engine valves. Test results verified the previous temperature rise data and showed little change with or without solvent flow.

The work described to this point was completed April 7, 1967. With the imminence of SC-3 launch date, an accelerated program was initiated to determine the stroke of a representative group of flight valves, and demonstrate by firing tests that the retroengine descent shutdown transients for the three engines would be within acceptable limits. At this time, the decision was made to send a group of flight valves to AFETR and determine the stroke of each at ambient temperature and at 165°F. The JPL stroke-measuring equipment was delivered to AFETR April 9. Fifteen flight valves were assembled from various sources; stroke measurements were completed by April 11.

The results of these stroke measurements, together with certain RMD acceptance test information for each valve, are presented in Table 15. As indicated, the strokes of the 15 valves range from 0.016 to 0.007 in. at ambient temperature. After a temperature soak at 165°F, the

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Table 15. Flight solenoid valve tests at AFETR

Engine serial No.	Solenoid valve SN	RMD acceptance test data		AFETR stroke, data in.			
		AT firing-shutdown impulse, lb-s	Closing[a] time, ms	Date, 1967	Ambient temperature	165°F	ΔStroke, amb to 165°F
549	213	2.16	47.6	Apr 10,	0.0160	0.0120	0.0040
547	230	1.89	39.6		0.0135	0.0090	0.0045
557	249	1.58	32.0		0.0150	0.0085	0.0065
558	222	1.51	34.0		0.0140	0.0080	0.0060
537	203	2.18	31.0		0.0115	0.0075	0.0040
562	242	1.76	—		0.0130	0.0075	0.0055
551	247	2.12	—		0.0115	0.0070	0.0045
552	229	1.36	—		0.0125	0.0070	0.0045
565	241	1.98	42.0		0.0115	0.0065	0.0050
536	244	2.24	44.0		0.0105	0.0055	0.0050
566	216	2.06	32.0		0.0100	0.0055	0.0045
555	235	2.00	—		0.0100	0.0045	0.0055
564	237	1.89	40.8		0.0080	0.0040	0.0040
560	248	1.82	34.0		0.0085	0.0020	0.0065
559	250	1.56	48.5		0.0070	0.0020	0.0050
563 (SC-3, leg 1)	224	2.16	35.0		—	—	—
554 (SC-3, leg 2)	236	1.67	47.4		—	—	—
556 (SC-3, leg 3)	245	1.75	37.0		—	—	—

[a] Ambient temperature flow bench data. Time interval from voltage signal to first sharp rise in oxidizer pressure at shutoff valve.

strokes range from 0.012 to 0.002 in. The average reduction in stroke from heating is about 0.005 in. No data could be obtained on the stroke of the three SC-3 solenoid valves, since propellant loading had been completed by this time.

With this information at hand, it was apparent that retroengine descent firing data were needed to determine the shutdown timing and impulse of an engine equipped with a solenoid valve whose stroke could be as low as 0.002 in. at 165°F. Accordingly SN 248, 250, 249, and 237 solenoid valves were shipped to ETS and preparations were made for a series of engine firing tests under simulated retroengine descent conditions. Of major concern in these tests would be the effect of solenoid valve temperature at the end of the firing cycle upon its closing characteristics. The final valve temperature depends on the amount of internal heat developed in its actuated condition, on the heat conduction to and from the propellants flowing through the shutoff valve, and probably to a lesser degree on its initial temperature (preconditioned to expected flight values or higher).

Five tests (DY-122 to -126) were performed from April 13 to 15, and they provided the data necessary to authorize the SC-3 launch on April 17. A sixth test (DY-127) was performed April 18. The tests were performed with a rebuilt TAT engine (SN 506). All tests were of 222-s duration and were performed with helium-saturated propellants. The firing profile was based on a HAC-supplied tape which included 55 s of constant thrust at "0" ΔmA, followed by the actual SC-1 (Leg No. 1 engine) terminal descent profile. Voltage applied to the solenoid valve was a constant 18.7 V dc. Provision was made for temperature conditioning the propellants and the engine valves. Two thermocouples were welded to the wall of each of the solenoid valves.

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Table 16. Solenoid valve firing tests at ETS (engine SN 506)

Test No.	Date, Apr 1967	Solenoid[b] valve SN	Data[a] time, s	Temperature, °F[d]							Shutdown data			Solenoid Stroke at 165°F, in.
				TO	TF	T-SOL-1	T-SOL-2	TFS	THE-2	TC-2	Thrust, lb	ISOV to start of F-decay, ms	Shutdown impulse, lb-s	
DY-122	13	250	0	76.5	86	119	118	117	113	110	—	—	—	0.002
			220	97.5	100	173	185	226	190	182	28.9	630	21.4	
DY-123	13	248	0	64	83	118	118	112	[c]	104	—	—	—	0.002
			220	93	96	178	189	236	[c]	181	29.0	∞	∞	
DY-124	14	250	0	69	71	102	102	101	102	101	—	—	—	0.002
			220	62	74.4	154	161	200	177	166	29.9	104	5.2	
DY-125	14	248	0	54	62	96	96	96	96	92	—	—	—	0.002
			220	56	66.5	145	149	176	156	157	31.0	66	3.4	
DY-126	15	249	0	70	54.5	97	97	98	98	96	—	—	—	0.0085
			220	76	52	153	162	210	178	163	28.8	30	2.4	
DY-127	18	237	0	69	62	98	97	95	98	93	—	—	—	0.004
			220	76	51	155	161	203	178	169	28.5	70	3.3	

[a] Nominal test duration—222 s.
[b] Nominal applied voltage to solenoid valve—18.7 V dc.
[c] Data channel malfunction at 7 s.
[d] Symbol definitions:
TO—oxidation temperature at throttle valve inlet
TF—fuel temperature at throttle valve inlet
T-SOL-1—solenoid barrel temperature
T-SOL-2—solenoid barrel temperature inlet end
TFS—fuel temperature at jacket outlet
TC-2—engine case temperature midpoint
THE-2—shutoff valve body temp
ISOV—current to solenoid valve

Table 16 presents all pertinent test data. The first two tests (DY-122 and -123) were performed under worst-case conditions, namely: (1) with minimum stroke solenoid valves (SNs 250 and 248); (2) 3 σ estimate of valve temperature at start of firing; and (3) 3 σ terminal descent duration. Preconditioning temperatures were about 20°F higher than expected on the mission. In the first test a very slow shutdown took place; in the second test the shutoff valve did not close, and shutdown was by emergency valves. Solenoid valve body temperature was about 190°F at shutdown.

Preconditioning temperatures were lowered to more realistic flight values in the second two tests (DY-124 and -125) and the same short-stroke solenoid valves were tested. A lower solenoid valve body temperature (about 160°F) permitted a reasonably short shutdown transient. Shutdown impulses of 5.2 and 3.4 lb-s, respectively, provide a measure of the valve action and may be compared with the RMD acceptance test shutdown impulses (column 3, Table 15), when using these valves (1.82 and 1.56 lb-s) in a 75-s ambient temperature acceptance test.

One of the longer stroke solenoid valves (SN 249) was attached to the engine for test DY-126. With a solenoid valve body temperature of about 160°F at the end of the test, shutdown was about twice as fast as with the short-stroke valve. Shutdown impulse was 2.4 lb-s.

The final test (DY-127) in the series was performed with SN 237 solenoid valve, whose stroke at 165°F (0.004 in.) was about twice that of the initial valves tested. Again solenoid valve body temperature at the end of the test was about 160°F; shutdown impulse was 3.3 lb-s.

It is apparent that flight qualified valves for future missions must meet more stringent requirements as to stroke. Criteria for flight valves are established with a prescribed stroke of not less than 0.0055 in. at 165 to 170°F.

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It is notable that SN 506 thrust chamber assembly successfully performed six simulated midcourse plus terminal engine descent missions with the development of only one minor leakage point (torque motor oxidizer omniseal) following the fourth test. An additional point of interest is the fact that the shutoff valve on this engine was equipped with the Fluorosint fuel poppet designed to meet lunar survival temperature extremes (SPS 37-43, Vol. I, p. 7).

f. Mectron filter tests. One of the topics considered by the SC-2 failure review board was the delay in shutdown of certain engines (i.e., burn No. 27 of the Leg No. 1 engine) and the possibility that particle contaminants within the helium actuation line could have entered the solenoid valve and affected its operating characteristics. A program to evaluate the effectiveness of a TRI-DOT filter (produced by Mectron Industries, El Monte, California) located at the helium inlet port of the shutoff valve was initiated. The filter configuration, filter sizes (10, 17, or 40 nominal micron rating) and possible influence of such a component on engine operability, is to be investigated.

The test work was initiated March 14, 1967 following procurement of eighteen units of different sizes and ratings. A standard "bubble test" was first performed to determine the absolute micron rating of the several filters. Following this test, 12 separate filters of three different ratings were mounted on three different shutoff valves and subjected to solvent flow bench tests (FB-192 to -194) to determine whether the presence of the filter affected valve opening response. No detectable delay was noted. Continuing, one filter of the finest rating was tested to determine whether isopropyl alcohol (in simulation of MMH • H_2O, which might be contained in the flight line) in the helium actuation line would affect opening response. Several actuation tests (FB-207 to -209) were performed with varying quantities of alcohol inserted in the line (0, ½, 1, 2 cm^3) upstream of the filter. A nominal 10-μm filter was used throughout this test series. Subsequent changes in the shutoff valve operating characteristics were determined. Table 17 summarizes the results and indicates that the presence of a filter has no effect on actuation times when the line is dry.

The response data from a similar test series (Test C-110 described in SPS 37-44, Vol. I, p. 25, and made with alcohol but no filter in the helium actuation line), when compared with the results of flow tests FB-207 to -209, indicate that the presence of a filter has no effect on actuation times when the line is wet. The response data for test C-110 are presented in Table 18.

Table 17. Actuation test series FB 207 to 209 results

Quantity of alcohol injected in helium line, cm^3	Presence of filter	Start time—VSOV[a] to PFSOV[b], ms	Shutdown time—VSOV to POSOV,[c] ms
0	Yes	14.0	18
0.5	Yes	15.0	21.5
1.0	Yes	18.5	25.5
1.5	Yes	25.0	32.5
2.0	Yes	17.0	22.5
0	No	14.5	19.5

[a] Voltage to shutoff valve.
[b] Pressure in fuel port of shutoff valve.
[c] Pressure in oxidizer port of shutoff valve.

Table 18. Test C-110 response data

Quantity of fuel injected into helium line, cm^3	Presence of filter	Start time—VSOV[a] to PFSOV,[b] ms	Shutdown time—VOSV[a] to POSOV,[c] ms
0	No	14.0	20.0
1	No	16.0	20.0
2	No	14.0	21.0

[a] Voltage to shutoff valve.
[b] Pressure in fuel port of shutoff valve.
[c] Pressure in oxidizer port of shutoff valve.

I. Digital Television Image Processing[3]

1. Introduction

The *Surveyor* digital television image processing is accomplished in the JPL image processing laboratory (IPL) facilities by digitally correcting and enhancing television pictures returned to earth by *Surveyor* spacecraft. The system may be divided into the following sections, each of which will be further discussed: interface; analog-to-digital conversion; preprocessing operations; picture processing operations; and output. A block diagram of the interface, analog-to-digital conversion and preprocessing operations is shown in Figs. 20 and 21 for flight data and calibration data inputs. The picture processing and output functions are given in Fig. 22, while Fig. 23 shows the picture processing operations in greater detail. The time indicated by each block is an estimate of through-put time for the particular function and for a typical run. For this article, a typical run is assumed to be eight 600-line TV frames spread over two FR-1400 analog tapes.

[3] Prepared by JPL Technical Section 324.

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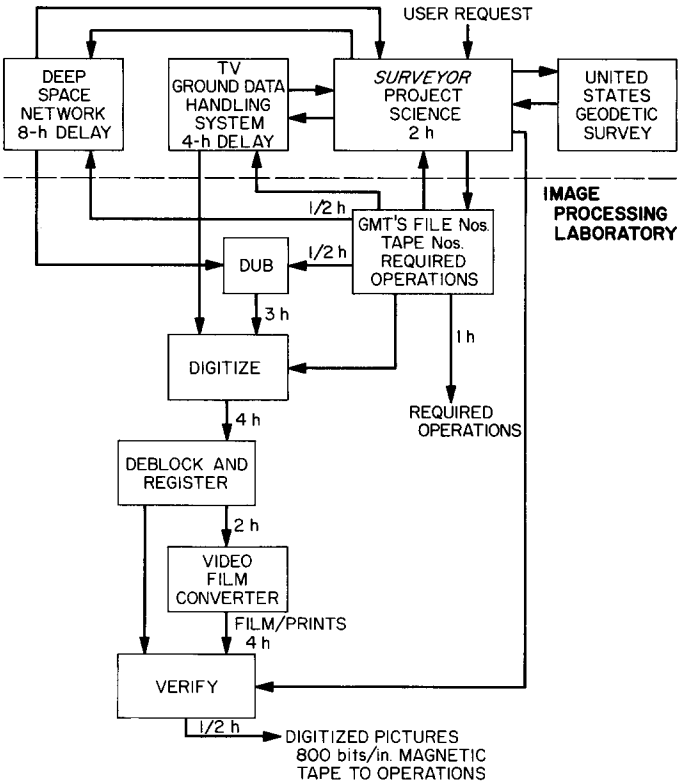

Fig. 20. *Surveyor* information flow: flight data inputs

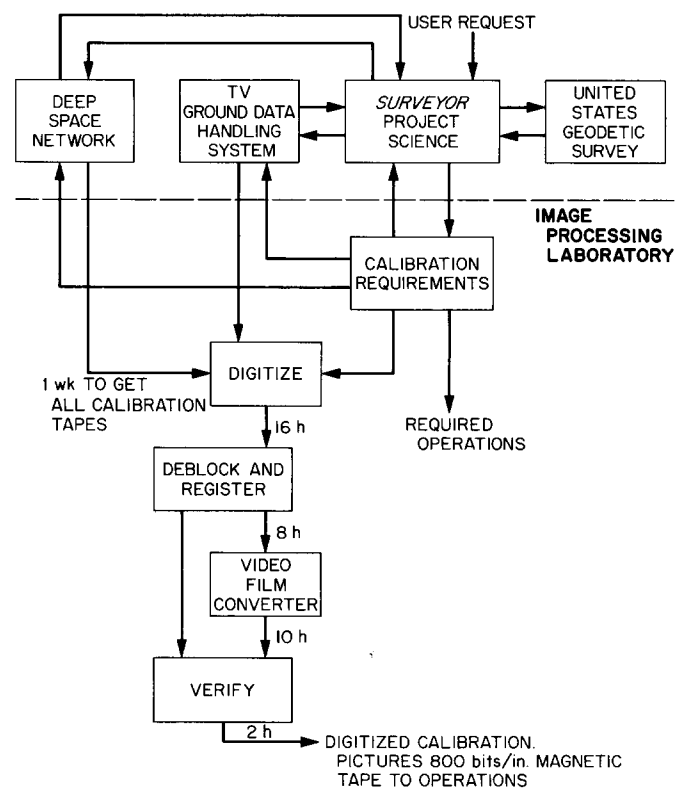

Fig. 21. *Surveyor* information flow: calibration data inputs

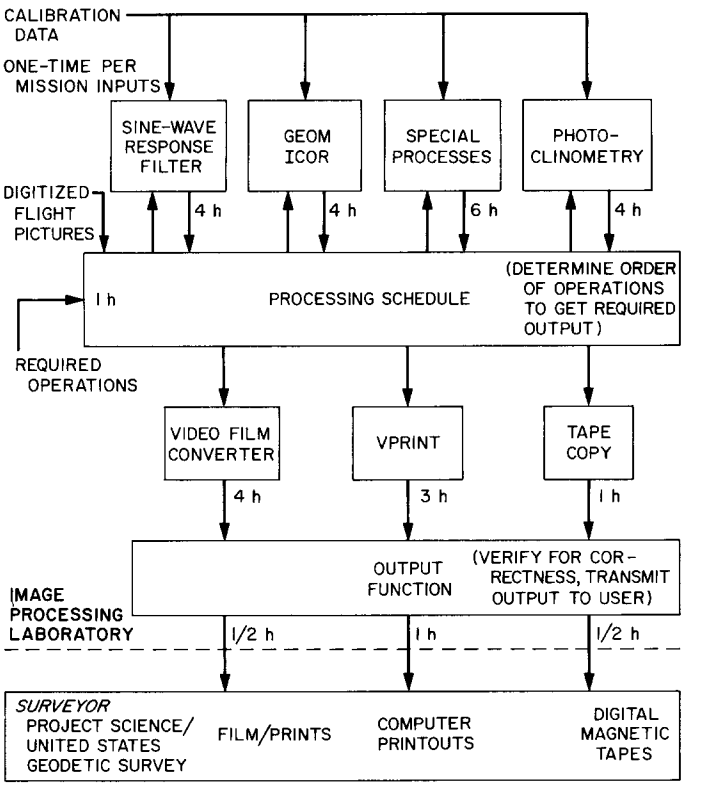

Fig. 22. *Surveyor* information flow: processing operations

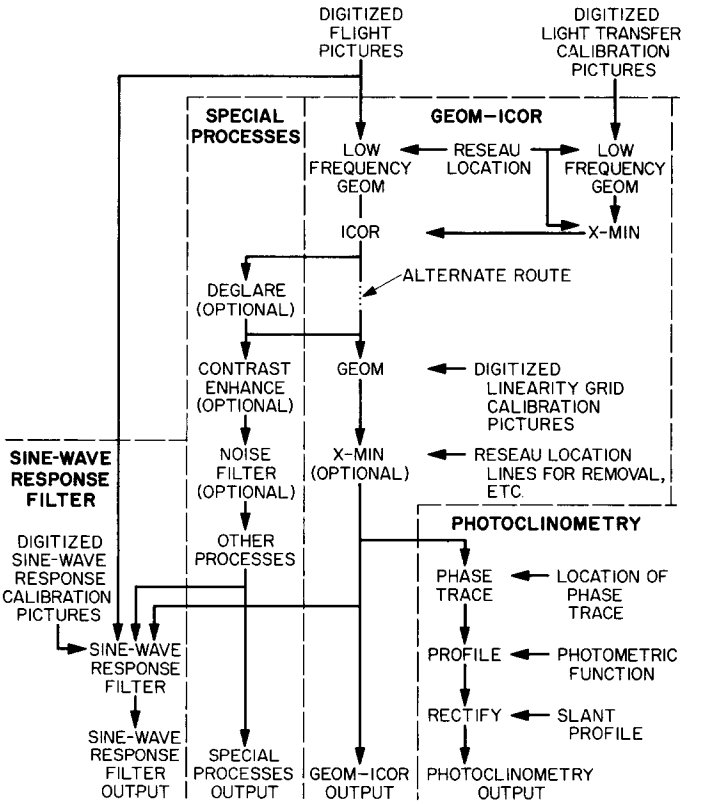

Fig. 23. *Surveyor* information flow: processing detail

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2. Interface

The interface between the IPL and the *Surveyor* project is the *Surveyor* television image processing cognizant engineer, through the science TV data engineer (STVDE). All requests for processing, priority determination, etc. are handled in this manner. The required tabular or positive enlargement output can be transmitted directly to the user; however, all film negatives are distributed only at the direction of the STVDE. Contact is also maintained with the TV ground data handling system (GDHS) and the Deep Space Network (DSN) [through Space Flight Operations Facility (SFOF) document control] for obtaining FR-1400 analog tapes containing the frames required for digitizing.

3. Analog-to-Digital Conversion

The first step in the *Surveyor* digital video processing is analog-to-digital conversion, shown in Fig. 20 as the "digitize" block. The required frames are digitized from the analog tapes and the digitized output recorded on digital magnetic tape at 800 bits/in. Depending on project requirements to preserve prime data, it may be necessary to dub analog tapes from the DSN stations and digitize from the dubs. The dubbing process adds about 3 h to the 4 h required for the actual digitizing of a typical run. The time required for digitizing depends heavily on the number of analog tapes involved. About 1 h/tape is spent in locating the frames and in general tape handling. The magnetic tape output consists of one long record for each picture with the pictures separated by file marks. Each 6-bit character of the record contains a digital representation of the TV camera's output. The analog-to-digital conversion is controlled by a 1-MHz reference tone recorded on the analog tape. Line sync is indicated by "all ones" (octal 77) characters on the digital tape.

4. Preprocessing Operations

Before any digital processing can be done it is necessary to change the analog-to-digital conversion output into a form more easily adaptable to the processing operations. This is done by Deblock and Register, a computer program which takes the digitized tape, finds line sync, and formats the picture into an 800-bits/in., 1-line/record, 1 picture/file tape, which then serves as the input to further processes. As part of the preprocessing operations a tape compatible with the video film converter (VFC) is made and the film is produced. This film is compared with analog check prints from the TV GDHS to verify the correctness of the digitizing and deblock and register operations. The times given are for a typical run of 8 pictures and are essentially directly proportional to the number of pictures. Figure 20 shows the preprocessing operations in block diagram form.

5. Calibration Data

The calibration data required for picture processing operations follows essentially the same path as the flight data through the analog-to-digital conversion and preprocessing operations, as indicated in Fig. 21. More time is involved in a calibrated run, however, since a typical calibration run is about 36 frames (e.g., light transfer characteristic—6 f-stops, 6 light levels/f-stop). Dubbing is not required for calibration tapes. In general, calibration data need be digitized only once per mission.

6. Picture Processing Operations

The major processing programs currently available or under development are: sine wave response filter (SWRF); geometrical distortion removal program (GEOM); intensity correction program (ICOR); and photoclinometry. In addition, special processes such as glare removal, and contrast enhancement are easily adaptable to *Surveyor* processing. Based on the requested processing, the processing schedule is set up. The overall data flow is given in the processing operations diagram (Fig. 22). All time estimates are based on the use of the IPL's IBM 360-44 computer. Those programs, which must, at present, use the 7094, will take 8 to 10 h longer, on the average. A more detailed flow diagram for each of the major processing programs is found in the processing detail diagram (Fig. 23).

The SWRF program restores high frequency information in both the horizontal and vertical directions in accordance with the response obtained from the calibration data. This program requires about 30 min/picture at present and uses the IBM 360-44 computer.

The GEOM and ICOR programs correct for geometrical distortions and variations in brightness sensitivity over the picture, due to the spacecraft camera system. The corrections are based on calibration data obtained from pictures of known geometrical shapes and light intensities. Application of both GEOM and ICOR to a picture takes about 15 min. These programs are now operational. They currently are run on the 7094 computer, but are scheduled for use on the IBM 360-44 by August 1967.

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The photoclinometry program currently under development with United States Geodetic Survey will provide corrected digital outputs of the intensity along a given direction within a particular frame. It is expected to take about 30 min/frame for the complete correction, once the techniques are worked out and programmed. Present scheduling calls for the program to be ready about October 1967.

Special programs for contrast enhancement, glare removal, coherent noise filtering, etc. are available. Due to the special setups required and the inherent cut and try nature of the methods used, the average run of 8 pictures should take about 6 h. Most of these programs may be run on the IBM 360-44.

In the processing detail diagram (Fig. 23) two previously undiscussed programs are shown. Low-frequency GEOM is a gross correction to be used primarily for adjusting flight and calibration pictures for the same reseau locations. They are generally different, due to changes in line rate between calibration and flight. X-min is a program which will effectively remove reseau marks, vidicon blemishes, dirt spots, or single lines of video and insert data for those elements by interpolation. This program should be ready for use on the IBM 360-44 by July 1, 1967.

7. Output Operations

The output from *Surveyor* digital video processing can be either film (35 or 70 mm), photo prints, computer printouts or digital magnetic tape (Fig. 22). Magnetic tape is generated by all the processing programs, and tape copying routines. Computer printouts of the digital value of all or of selected picture elements within a frame can be obtained through a digital value printout routine (VPRINT). Until about June 30, 1967 this must be done on a 7094 computer. The computer time involved is minimal (5-10 min/run), but the printer takes the majority of the 3 h estimated. On the 7094 an average delay of at least 1 day may be expected. From the processed output tape, a tape compatible with the VFC is produced by the VFC mask program. This tape is played back through the VFC to produce film. The processed film and/or prints then become the output. This whole process requires about 4 h/run.

8. Processing Capacity

The present processing capacity of the IPL for *Surveyor* pictures is 25 pictures/wk. This is based on a single 40-h shift operation. The assumptions and reasoning involved in the estimate of this capacity are outlined below.

(1) The overall digitizing function (dub, digitize, deblock and register, VFC, and verify blocks on the block diagram) can be performed in parallel with the processing (SWRF or GEOM and ICOR or photoclinometry or special processes) and output (VFC, etc.).

(2) The digitizing function requires about 12 h/8 pictures, giving a rate of 5 pictures/day.

(3) Processing and output can proceed at the rate of 5 pictures/day using the IBM 360 computer. Special processing will be slowed to about 4 pictures/day.

(4) Assumption: No processing will be done on the 7094 computer except possibly VPRINT.

(5) Assumption: All processing will require film output.

(6) Assumption: Special processing such as deglare, contrast enhancement, etc. will be needed for a maximum of 20% of the pictures.

(7) Assumption: All work is divided into runs of about 8 pictures each.

(8) The stated weekly capacity is a long-term average (1 mo or more). Due to delays in getting required tapes, capacity probably cannot be reached in < 2 wk.

(9) Support during the mission operations period cannot be given at the stated level because of the recall of technicians to SFOF.

9. Hardware Status

The following hardware in the IPL is used to support *Surveyor* digital video processing, although *Mariner* and advanced development funds are being used in addition. A brief status report is included.

(1) Computer—An IBM 360 Model 44 digital computer was delivered by March 15, 1967 and is currently operational for most *Surveyor* processing. Minor problems of tape compatibility with the video tape converter (VTC) were encountered, but these have been rectified.

(2) VTC—This equipment is used to digitize the *Surveyor* TV picture directly from the analog tape and has been operational since about February 15, 1967.

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(3) VFC—This equipment is used in *Surveyor* processing to produce 35- or 70-mm film negatives from computer-generated magnetic tapes. Analog magnetic tape may also be used. While the VFC is operational, continuing improvements will be made, including character generation and interfacing with the IBM-360 computer.

(4) VTC translator—This equipment is currently being checked out. At present 200-bits/in. tapes can be played back about three times faster than with the old system. When checkout is complete it will provide the capability of playing back digital tapes at 800 bits/in. to produce film in the VFC. It will also enable the use of 8-bit data in addition to the present 6-bit data as input to or output from the VFC.

10. *Surveyor I* Picture Processing

As of June 15, 1967 a total of 186 *Surveyor I* 600-line TV pictures had been digitized. Of these, 103 have been filtered with the SWRF program and are essentially completed. Minor work involved in getting a final 35 mm film output for 34 of these frames remains and will be completed by June 30, 1967. Seventy-five frames have been digitized in order to obtain computer printouts of selected portions of each picture and in some cases the entire frame. All the required printouts have been delivered, and the work remaining for the final 35 mm film output for 17 of these frames will be completed by June 30, 1967. Of those pictures still in process three are ready for ICOR processing, as described elsewhere in this article, and five more are in the final stages of SWRF processing. Processing should be completed by July 15, 1967.

Five requested pictures cannot be digitized. One or two may possibly be recovered from the original FR-1400 or an FR-1400 dub of the on-site FR-700/800 recordings; however, this is not presently planned.

Considerable work with the *Surveyor I* calibration data has been performed. About 250 calibration pictures have been digitized, including those necessary for the SWRF, ICOR and GEOM programs. The creation of the actual calibration files for the ICOR program remains to be done for the red and blue filters, however. This should be accomplished by about July 5, 1967.

11. *Surveyor III* Picture Processing

As of June 15, 1967 digital processing had been requested on 143 of the 600-line TV pictures from *Surveyor III*. Fifty-nine frames have been filtered with the SWRF program and are complete. An additional 25 frames have been digitized for computer printouts of selected portions or complete pictures. These have been completed.

As a result of the large amount of glare present in the *Surveyor III* pictures, a "deglare" program originally developed for IPL medical image processing work was adapted for *Surveyor* pictures and demonstrated on one of the flight frames. This program essentially removes low frequency data from the pictures and adjusts contrast. Work orders have been received to deglare 59 frames. About 30 of these frames are in process; the others are waiting to be scheduled. These frames are to be redigitized to maximize the surface detail before applying the deglare and SWRF programs.

Calibration work has consisted of the following digitizing: horizontal and vertical sine wave response — 18 frames; linearity—2 frames; colorimetric chart—3 frames; light transfer filters (red, blue, green) — 24 frames; and light transfer clear filter — 55 frames. Parameters for the SWRF filter have been determined and work should begin by July 15, on creating the actual calibration files used in the ICOR program (for red, blue, green, and clear filters). Printouts were obtained of both the photometric chart and light transfer characteristics.

J. Thermal Engineering

SC-5 solar thermal vacuum testing was performed. Anomalous behavior of the nitrogen tank and the helium relief valves during testing was investigated and satisfactory design changes incorporated.

During solar thermal vacuum testing the alpha scattering instrument sensor head ran 30°F warmer than predicted. The discrepancy has not been resolved as yet, but no operational problems are foreseen.

All other components performed nominally during the test phases.

A thermal analysis to evaluate the effects of new paint patterns to increase TV viewability for landing gears 2 and 3 indicates that attitude jets 2 and 3 may be caused to run 20°F higher during transit.

A study to evaluate *Surveyor III* compartments A and B thermal switch status at shutdown indicated that a maximum of only two switches on compartment A and three

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on compartment B were open at temperatures which should have opened all 15 switches.

K. Deep Space Network

1. Mission C Operations

Revival attempts were begun from deep space station (DSS) 61 at 13:30 PDT May 23, 1967 and were performed continuously until June 1. No response was received.

2. Modification Kit Installations

Command and data handling console (CDC) modifications were completed at DSSs 11, 42, 51, 61, 71, and 72. Modifications included those that satisfy the ground rule of keeping all deep space instrumentation facility (DSIF) CDC units as interchangeable as practicable. A total of 38 modification kits were installed at each of the three stations between April 1 and June 1.

3. CDC Maintenance and Operations Transfer

Updated open item sheets (list 11) were issued monthly to document progress toward the eventual closeout of the open items given in the two basic selloff plans: CDC Selloff Plan, DSIF-42 (3023338) and CDC Selloff Plan, DSIF-51 (3050571).

4. CDC Patchboard/Overlay Control

New overlay scales required for Mission D are in work. Revision G of CDC Decommutator Patchboard/Telemetry Overlay Control Specification 3023300 has been prepared. This revision incorporates recent changes to Spacecraft Data Channel Assignment Specification 3023904.

5. Development Tests

The following development test request bulletins were issued during this report period:

(1) DTRB 21—strain gage multiplex test. This test was completed and the report written and distributed.

(2) DTRB 23—alpha scattering downlink PCM telemetry test (short and long loop). This test is approximately 80% complete.

(3) DTRB 24—spectral analysis test of command SCO. This test was performed at Goldstone with no spurious emissions noted. The report has been written and distributed.

6. DSIF Detailed Operating Procedures

Numerous changes to the DSIF detailed operating procedures (DOPs) have been or are in the process of being implemented.

7. *Surveyor* Operations Chief Handbook

Revision D of SSD 68089R, "Surveyor Operations Chief Operational Aids Handbook," has been completed, with publication and distribution expected by June 15.

8. DSN Training

Training of Deep Space Network (DSN) personnel for their eventual assumption of CDC maintenance and operations responsibility from HAC continued.

9. Special DSIF Station Test, Operations, and Training

a. Development tests. The Madrid rerun of development test request bulletin 18, CDC/FR 800 interface test, was completed. The report was written and distributed.

b. New work—CDC 10/12 add-on equipment. Except for the two S-band signal sources, all the CDC 10/12 add-on equipment units and assemblies have been delivered to the DSN for checkout and shipment to DSSs 61 and 72, in accordance with the contract.

10. Trouble and Failure Reports (TFRs)

The breakdown by location of the TFRs remaining to be closed by HAC are as follows:

Goldstone	30
Canberra	13
Johannesburg	9
Madrid	10
Air Force Eastern Test Range	11
Ascension	3
System Test Equipment Assembly 1 (STEA 1)	43
STEA 2	28
STEA 3	17
STEA 4	38
STEA 5	20
STEA 6	15

There are four critical TFRs on the critical list at the completion of this report period. Engineering Change

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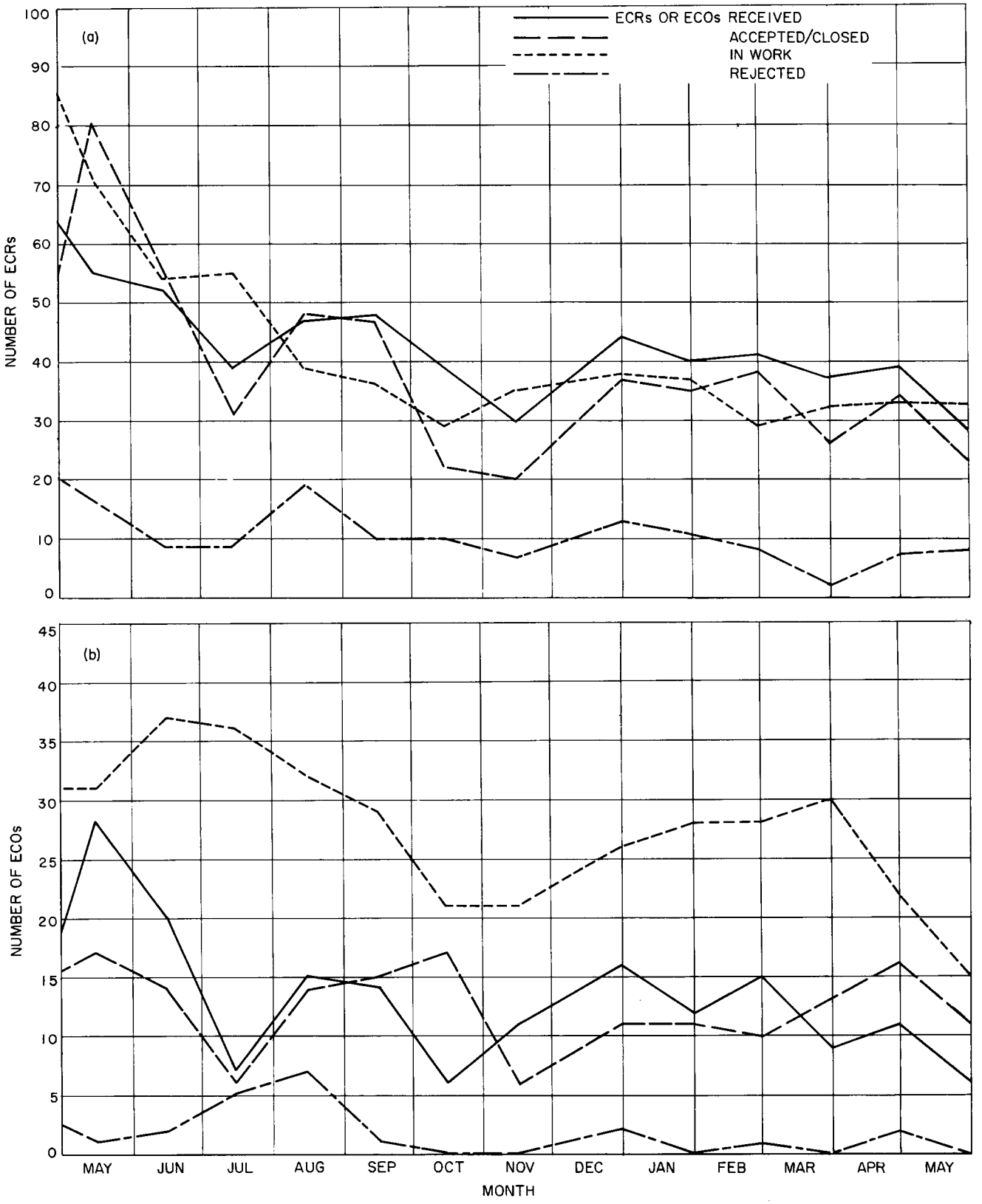

Fig. 24. Operational support equipment status and trends: (a) ECRs; (b) ECOs

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Authorizations (ECAs) have been released and modification kits sent to the stations. All these critical TFRs will be closed by July 1, 1967.

11. Modification Kit, Engineering Change Requests (ECRs), and Engineering Change Orders (ECOs) Status

Detailed status on specific ECRs and ECOs is given in the *Surveyor* project management plan, document 3023333, which is updated every 2 wk, and in the engineering change management report, also updated every 2 wk.

a. Modification kits. All modification kits authorized for installation in the DSIF CDC equipment during the first lunar night after mission C have been shipped, installed, checked out, and tested. Additional modifications have been approved for installation prior to mission D in the first 3 wk in June. Both lists of approved modifications will be installed at DSS 72 (Ascension) when HAC personnel return to the station.

b. ECRs and ECOs. Figure 24 shows the status and trends of ECRs and ECOs for this and past report periods. The average ECR and ECO work loads have declined to about 32/mo and 18/mo, respectively.

12. Operation and Maintenance Manual Upgrade

A number of interim data sheets and formal change notices were prepared and distributed during this reporting period. A priority listing of all CDC equipment items requiring operation and maintenance manual rework is being prepared. Following assignment of firm priorities, a master program and schedule will be prepared.

13. New Work—Alpha Scattering

First phase performance testing has been conducted at Goldstone using the new alpha scattering CDC equipment with results indicating a possible 2-dB margin better than specification requirements.

14. CDC STEA Configuration Accounting

A CDC STEA configuration accounting plan has been prepared which sets forth the procedure to gather exact information concerning the engineering configuration of the CDC STEA equipment. When gathered, the information will be disseminated in the form of officially released CDC STEA equipment configuration indexes. Upon approval, implementation of the plan will begin immediately.

L. Quality Assurance and Control

1. Disengagement Force Measurement

Modification of the SQAD that presents disengagement force requirements was necessary to comply with an SC-3 consent-to-launch action item. All required modifications have now been generated and clarified by JPL and HAC. Presently being revised and implemented is SQAD 18-12.

2. Quality of Product

a. Radar doppler velocity sensor (RADVS) signal data converter (SDC). The RADVS SDC, serial number (SN) 5, had solder joint failures. A failure review board was established to investigate soldering problems at Ryan Aeronautical Company, San Diego, California. As a result, all solder joints that could be inspected without disassembling modules from their motherboard were reinspected and those found questionable were reworked.

Corrective actions were as follows: (1) supplier notification; (2) motivation program for soldering personnel; (3) intensified inspection program by both the supplier and HAC source inspection, using magnification; and (4) engineering evaluation of environmental factors which may be contributing to the problem.

b. Electrical connectors. The 22-gage female sockets of electrical connectors manufactured by Microdot Inc., South Pasadena, California, experienced a high failure rate during disengagement force testing. The problem was caused by the tweezers used to hold the socket while it was being soldered to associated wiring. The contact spring was being deformed in a manner that made it break when subjected to subsequent engagements.

This problem has been isolated to:

(1) Those 22-gage female sockets that are the latest of two specific designs. The old design provided socket contact by slotting the barrel of the female socket; the new design uses a pencil-clip type of contact spring inserted through a slot cut into the side of the socket barrel.

(2) Assemblies containing such new design connectors are limited to those whose internal wiring is soldered to female connector pins. Specifically, the flight control sensor group (FCSG) and the TV camera.

With the exception of the TV cameras for SC-4 and SP-1, all cameras have been inspected and determined to be manufactured with female sockets of the old design, which are not susceptible to this problem. The SP-1

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camera was found to contain the old socket design on the camera and the new one on the harness that goes from the mirror assembly to the electronics assembly. The SC-4 camera will either be inspected or X-rayed.

With the exception of SC-4 and -5, the FCSG sockets are being changed to the old female socket design. SC-4 and -5 FCSGs will undergo special handling.

To identify the cause of the problem, plus some other potential areas of concern, SQAD Bulletin 0.117 has been issued. It also requires that certain precautionary measures be implemented to preclude recurrence.

c. Spacecraft installation hardware. There has, on occasion, been a problem in the spacecraft areas when excess or loss of installation hardware (nuts, bolts, screws, etc.) has been evident. Spacecraft installation hardware and tools are required to be accounted for, both before and after installation, by SQAD 20-9.

d. Control items. Control item manufacturing departures are being detected in the control items delivered to the spacecraft. An investigation resulted in the determination that deficiencies known to spacecraft level personnel as potential hazards to the spacecraft were not being consistently reported back to the appropriate unit areas. Actions implemented were:

(1) A list of problem areas generated by spacecraft personnel is used as an inspection checklist in the unit areas. This list will be maintained as experience dictates.

(2) Items to be installed on the spacecraft will undergo a review by unit area and spacecraft supervision before installation.

(3) Unit area supervision will perform a hardware review following the final inspection point.

M. Radar Altimeter and Doppler Velocity Sensor (RADVS)

1. RADVS Performance Evaluation During *Surveyor III* Lunar Mission

Examination of mission C flight data shows that RADVS contributions to the descent flight path were within planned sequence tolerances until shortly after the spacecraft reached a descent velocity approximating 5 ft/s at an estimated altitude of 34 ft above the lunar surface. At this point, RADVS beam-3 broke lock in its automatic tracking of the surface referenced doppler velocity signal. This, in turn, resulted in loss of the reliable operate radar altimeter and reliable operate doppler velocity sensor logic signals, and the spacecraft flight control system was immediately commanded back to the inertial reference unit sensor. This leaves the minimum acceleration command to control vehicle velocity. Injection of RADVS signals into the vehicle flight control is shown in Fig. 25.

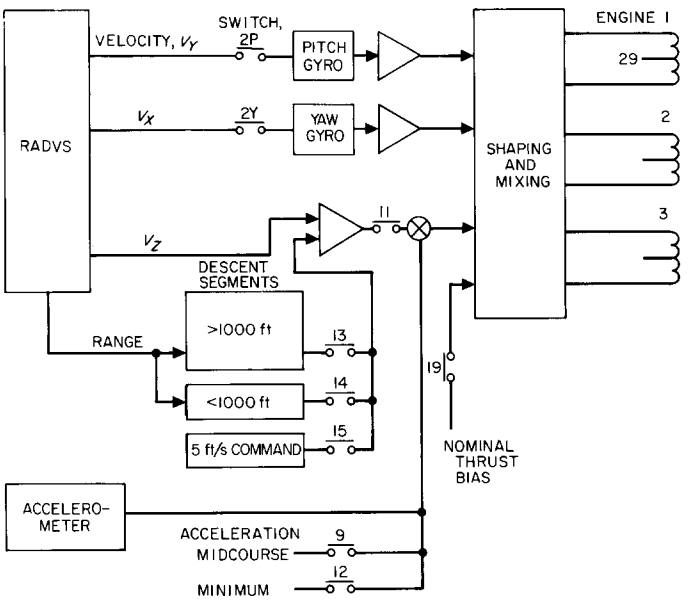

Fig. 25. Simplified flight control diagram

Loss of lock by the frequency D3 tracker (beam-3) occurred during preamplifier gain switching. Telemetry data show that all three doppler velocity sensor preamplifiers were solidly operating in gain state 1 from the 1000-ft mark until approximately 3 s prior to the 10-ft/s signal. It was at this point that the preamplifier gain state switching occurred, showing a significant decrease in surface reflected radar signal, a condition not experienced during mission A. Preflight analyses predicted an 18-dB margin against a doppler velocity sensor beam ever getting into the highest gain state (90 dB) in the critical region that occurs at first contact with the 5-ft/s vertical velocity.

Figure 26 indicates that received power for all beams was lower than predicted throughout the terminal descent. During descent portions for which gain state switching was at a minimum, reflectivity data for each of beam-1, -2, and -3 indicate received power to be generally 2 to 4 dB below that expected. Beam-4 reflectivity indicates from

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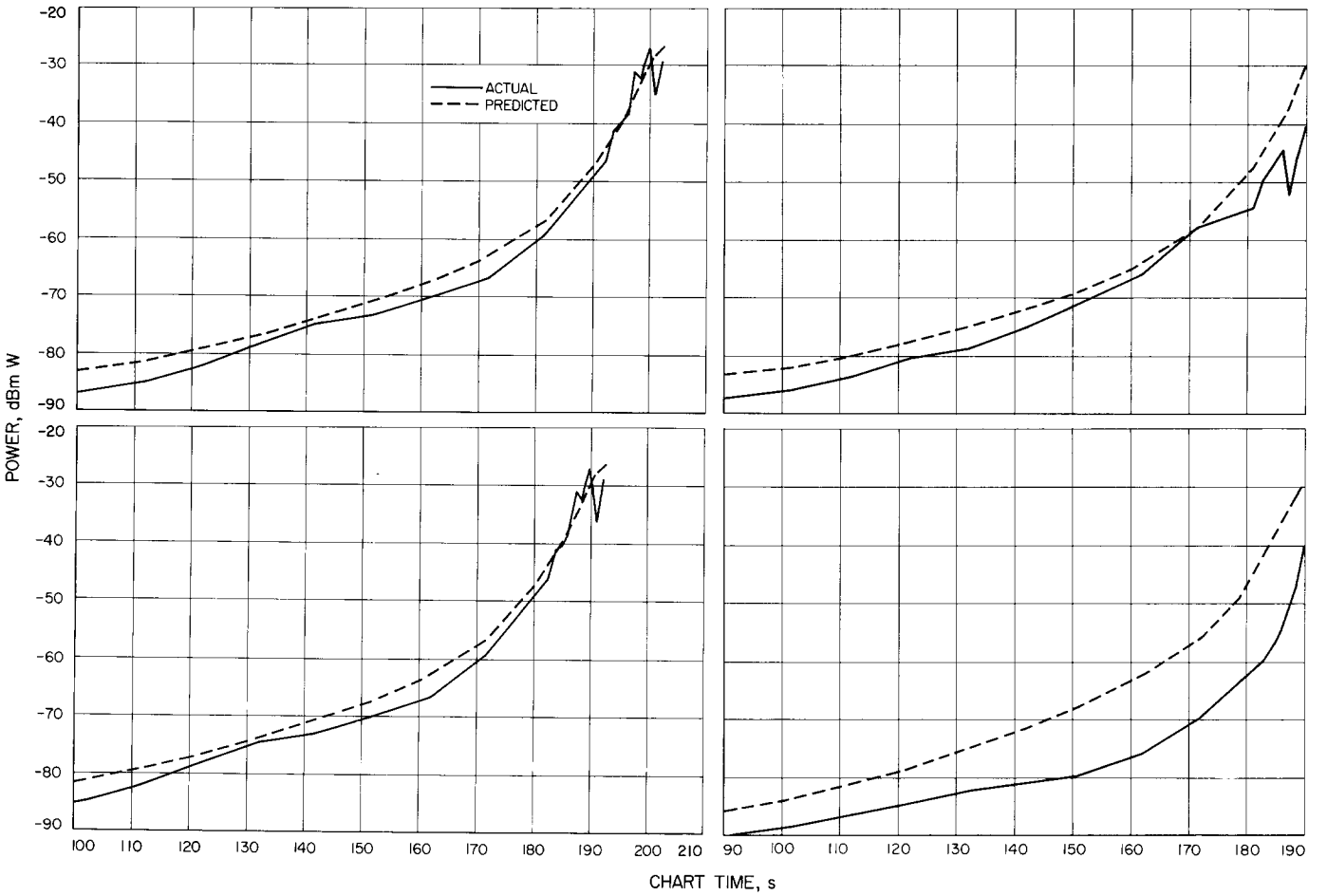

Fig. 26. Received power vs predicted

4 to 19 dB less received power than predicted.[4] Although all beams are approximately 4 dB below predictions early in the terminal descent (at a 25-deg angle to the spacecraft Z-axis) the return power of the doppler velocity sensor beams generally approaches the prediction more closely with decreasing slant range. Beam-4 (along the spacecraft Z-axis), however, departs from the prediction with a decreasing slant range to a maximum of 19 dB at a slant range of approximately 60 ft. It appears quite probable that the sloped terrain of the touchdown point and the decreasing area of radar illumination with decreasing slant range caused this phenomenon.

Since there is no evidence of RADVS subsystem malfunction in the available data, the cause of the signal fade and the related receiver-transmitter preamplifier gain state switching is probably attributable to changes in lunar terrain detail. At the time of unlock, the power levels at the receive feed horns appear to be stronger than −35 dBmW into beam-2 and stronger than −75 dBmW into beam-3. In view of signal strengths of these magnitudes, it appears very probable that the beam-3 unlock was caused by the crosscoupled sidelobe logic circuitry.

Under the operating conditions existing at this point of *Surveyor III's* descent trajectory (approximately 34 ft above the lunar surface), all four criteria for "break lock" on beam-3 were met, causing the D3 frequency tracker to respond as designed and revert to a normal search mode. Since the search sweep range of these trackers does not go below 800 Hz, reacquisition at the then achieved vertical descent rate (approximately 5 ft/s) is not expected unless the doppler frequency amplitude is significantly greater than normal, a condition which did not exist.

[4]In Fig. 26, the beam-4 reflectivity actual plot represents considerable smoothing and averaging. Because of this, an accurate interpretation of the data is difficult. (The Ryan Aeronautical Company interpretation indicates less dispersion between the actual and predicted curves.)

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a. Crosscoupled sidelobe logic modifications for SC-4 and subsequent spacecraft. A design change has now been incorporated in SC-4's RADVS signal data converter which will inhibit active response of the crosscoupled sidelobe logic circuitry below 1000-ft altitude, i.e., after the 1000-ft mark. This change will in no way compromise spacecraft flight performance in that the functional requirements for an active crosscoupled sidelobe logic system exist only during the early part of the descent trajectory, or prior to reaching the vertical portion of the descent path. From this point on in the trajectory, as a sidelobe is received by an antenna, the mainlobe is superimposed and is much stronger, so no deleterious effects are experienced. In the event of any short-term mainlobe dropout, even if a sidelobe is briefly acquired, relock to the mainlobe will occur as soon as it reappears. The resultant effect is that sidelobe logic is reacquired for only the first 10 to 30 s of radar steering. While the 1000-ft level is substantially beyond this basic crosscoupled sidelobe logic requirement, it is the first available signal generated within the signal data converter which can be used for the desired crosscoupled sidelobe logic inhibit function. The 1000-ft mark sets the disable input to the inhibit gate to a logical zero, thereby causing the frequency comparison input to the crosscoupled sidelobe logic end gate to be a zero, and thereby inhibiting crosscoupled sidelobe operation.

Because of the complexity of this logic (some 700 to 850 components), it appears that it would be desirable to add a telemetry signal to monitor a forced break lock. Two spare channels are available on word 12 in mode 6. A circuit addition of a gate in the signal data converter and wiring changes to both the converter and spacecraft harness would allow the monitoring of the three forced loss signals on a single telemetry channel. This would be satisfactory, since other data, such as tracker lock signals, would identify the channel that forced it into search. The feasibility and advisability of making this change will be further pursued with the hope of incorporating it into later spacecraft.

b. Klystron power supply modulator power interruption effect profile study. A brief study was conducted to establish the maximum time klystron power supply modulator primary power may be interrupted without causing a recycle of the klystron high voltage delay control circuitry. Under normal RADVS system operation, when primary power is applied to the starting circuits, filament voltages are applied to both klystrons through the filament converter and the two filament regulators. A time delay circuit provides a 20-s warmup period between application of filament voltage and the high voltage to each klystron.

The test data taken during this study show that primary power interruption periods greater than 2.85 and 2.77 s at 25 and 0°C, respectively, will cause the klystron power supply modulator to assume a recycle mode of approximately 20 s. If interruption periods are less, the klystron power supply modulator will not recycle, but the recovery time approaches 217 s, which is slightly greater than that required for the search delay of the signal data converter doppler tracker.

2. Failure Review Board

As a result of several apparently random failures of discrete components in the RADVS signal data converter, which occurred during flight acceptance testing after the crosscoupled sidelobe modification, a failure review board was organized to investigate the cause and to determine the corrective action required. Membership was drawn from the engineering reliability groups of Ryan Aeronautical Company, HAC, and JPL, as well as HAC *Surveyor* systems engineering and quality assurance.

Table 19 represents a summary of the failure review board recommendations.

Table 19. Recommendations

Trouble failure report (TFR) Number	Description	Recommendations
46671	SDC SN-11 L-1 failure	Replace L-1 with available spare Use all 700 series flight items as is Add sleeving to future build (10124[1]) indicators
46661	SDC SN-5 T-1 failure	Replace T-1 with available spare Use all 700 series flight items as is Increase terminal 16- and 17-wire size in future builds
46672	SDC SN-11 preregulator	Close TFR—as random pre-FAT failure
46663	SDC SN-5 V_s converter	Converter replaced—close TFR Establish C-lien on all spacecraft
46666	SDC SN-8 doppler velocity sensor tracker module	Module replaced—close TFR Establish C-lien on all spacecraft

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In addition to recommendations on an individual failure basis, the failure review board was directed to make recommendations on the flight worthiness of each affected signal data converter. The units directly affected are SNs 5, 11, and 8. The other flight units are affected because they have similar parts or history.

With the replacement of the signal data converters with later assemblies from Y-2 and after repair of the trackers and converters and completion of flight acceptance testing SN 5 will be considered a flightworthy spare. The other converter units will be considered flightworthy upon completion of the following actions:

(1) SN 7—Reinspection of all visible joints, rework, and retest for ECP-23 modification.

(2) SN 8—Reinspection of all visible joints and rework as necessary.

(3) SN 11 and SN 12—No action required.

Life span information with visibly defective solder joints to obtain data on expected life is highly desirable and is being gathered in tests currently being conducted at Ryan Aeronautical Company. These tests will be continued. A C-lien should be levied against all spacecraft based on RADVS solder joint uncertainty so that adequate management consideration will be given this item at consent-to-meetings. All present and any new solder joint TFRs should be closed out on an individual TFR basis, except for the generic implications which are covered by the recommended C-lien against all spacecraft. A more inclusive solder joint study investigation based on all equipments should be instituted as a separate *Surveyor* task.

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